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PERFORMANCE AND LOAD-RANGE CHARACTERISTICS OF TURBOJET
ENGINE IN TRANSONIC SPEED RANGE

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SUMMARY

In order to determine the optimum combination of compressor pressure ratio and turbine-inlet temperature on the basis of load range for flight speeds in the transonic range, an analysis was made of the performance and the load-range characteristics of the turbojet engine for flight speeds from 500 to 800 miles per hour, altitudes from 10,000 to 70,000 feet, compressor pressure ratios from 2 to 30, and turbine-inlet temperatures of 1700°, 2000°, and 2300° R. The values of the lift-drag and structure-to-gross-weight ratios of the aircraft and the efficiencies of the engine components assumed for this analysis are representative of the best values obtained either in practice or in laboratory investigations.

The variation, with flight conditions and engine operating variables, of the thrust per square foot of engine frontal area, specific weight, thrust specific fuel consumption, ultimate range, and range with pay load are discussed.

The following results were obtained for the case of a submerged engine installation: Maximum, or near maximum, ultimate range was attained at any of the flight conditions investigated with a compressor pressure ratio of about 8 to 10. For all speeds investigated at altitudes of 10,000 and 30,000 feet, and for speeds up to 700 miles per hour at 50,000 feet, the variation of ultimate range with turbine-inlet temperature within the temperature range investigated was about 5 percent or less, with a temperature of 1700° R giving the longest range at the lower speeds and altitudes. At an altitude of 50,000 feet and a speed of 800 miles per hour, and at all speeds investigated at 70,000 feet, a turbine-inlet temperature of 2300° R gave a 10 to 30 percent longer range than the temperature of 1700° R and a 4 to 10 percent longer range than the temperature of 2000° R.

INTRODUCTION

A theoretical analysis to provide an insight into the potential aircraft range and the most suitable operating conditions for six types of propulsion system is presented in reference 1. The engine types considered are: compound, turbine propeller, turbojet, turbo ram jet, ram jet, and rocket. Emphasis is placed on the maximum range an aircraft powered by each of the engine types could attain with no pay load.

In order to extend the investigation of reference 1 to a region of particular interest, an analysis of the turbojet engine in the region of transonic flight speed was made at the NACA Lewis laboratory to determine the optimum combination of compressor pressure ratio and turbine-inlet temperature on the basis of aircraft load-range performance. The performance of the turbojet engine and the load-range characteristics of aircraft powered by the turbojet engine were calculated for flight speeds from 500 to 800 miles per hour and for altitudes from 10,000 to 70,000 feet; reference 1 covers flight speeds above 800 and below 500 miles per hour. Additional calculations of the load-range performance of the turbojet engine for flight speeds lower and higher than 500 to 800 miles per hour were made to compare the load-range performance in the transonic-speed region with the load-range performance at other flight speeds. The compressor pressure ratio for the turbojet engine, which is optimized on the basis of thrust in reference 1, was optimized on the basis of aircraft load range herein. The turbine-inlet temperature, which is assumed constant at 2000° R in reference 1, was varied from 1700° to 2300° R. Several of the assumptions of reference 1 as to component efficiencies, lift-drag ratio, engine weight, and so forth have been changed in view of more recent information obtained in laboratory investigations.

METHODS

A diagram of the turbojet engine assumed for the analysis is shown in figure 1. The performance of the turbojet engine is calculated for flight altitudes of 10,000, 30,000, 50,000, and 70,000 feet; for flight speeds of 500, 600, 700, and 800 miles per hour; for compressor pressure ratios from 2 to 30; and for turbine-inlet temperatures of 1700° , 2000° , and 2300° R. Additional calculations, at flight speeds from 0 to 500 and from 800 to 1400 miles per hour, are made for an altitude of 50,000 feet. The turbojet compressor was assumed to be of the axial-flow type for pressure ratios up to 10. For pressure ratios greater than 10, the compressor system was assumed to be made up of two parts: an axial-flow compressor with a pressure

ratio of 10, followed by a centrifugal-flow compressor with whatever pressure ratio was required to achieve the over-all pressure ratio desired. The engine performance is calculated using the thermodynamic data of references 2, 3, and 4 for the compression, combustion, and expansion processes, respectively. The pressure drop in the combustion chamber was neglected, as it was found that the pressure drop was small and had a negligible effect on the engine performance.

The following constant quantities were assumed:

Axial-flow-compressor polytropic efficiency (total-to-total) .	0.88
Centrifugal-flow-compressor adiabatic efficiency (total-to-total)	.80
Combustion efficiency.	.98
Turbine adiabatic efficiency (total-to-total).	.90
Exhaust-nozzle velocity coefficient.	.97

The pressure-rise recovery factor (ratio of actual pressure rise to theoretical pressure rise) of the engine-inlet diffuser was assumed to vary with flight Mach number, as shown in figure 2. The curve of figure 2 shows representative values obtained from a survey of current literature.

The air flow for the turbojet engine was assumed to be 13 pounds per second per square foot of engine frontal area for sea-level zero-ram conditions at the compressor inlet. At other altitudes and flight speeds, the air flow was calculated by assuming that the axial Mach number at the compressor inlet remained constant at the value corresponding to the sea-level static air flow.

The weight per square foot of engine frontal area of the turbojet engine was assumed to increase with increasing compressor pressure ratio and turbine enthalpy drop. For corresponding compressor pressure ratios, the specific engine weights found using this assumption approximate the specific weights of the lightest of current engines.

The following assumptions of airplane characteristics were made: (1) structure- to-gross-weight ratio, 0.4; and (2) fuel-tank- to-fuel-weight ratio, 0.05. It was assumed that the maximum lift-drag ratio that the airplane could attain from pure aerodynamic considerations, regardless of gross weight, varied with flight Mach number, as shown in figure 3. The lift coefficient at the maximum lift-drag ratio, which is necessary for the calculation of the wing loading, was assumed to vary with flight Mach number, as shown in figure 4. The curves of figures 3 and 4 were obtained from a survey of the literature in this field. These assumed airplane lift-drag ratios and corresponding lift coefficients were used at all flight conditions for which they

did not result in a wing loading higher than 125 pounds per square foot. If the wing loading resulting from these assumptions was higher than 125 pounds per square foot, the airplane lift-drag ratio and corresponding lift coefficient were reduced by use of representative lift-drag polars for the different speeds so that the wing loading remained constant at 125 pounds per square foot. The following final values of lift-drag ratio were used:

Flight speed (mph)	Lift-drag ratio			
	Altitude, (ft)			
	10,000	30,000	50,000	70,000
100-400			20.0	
500	16.8	20.0	20.0	20.0
600	14.0	17.4	15.3	15.3
700	9.0	11.7	11.4	11.4
800	7.0	10.0	11.0	11.0
1000			9.5	
1400			5.5	

The engines were assumed to be submerged in the rear of the fuselage and consequently there was no nacelle drag, except as noted in the following paragraph.

Additional calculations were carried out for altitudes of 30,000 and 50,000 feet at speeds of 500 and 800 miles per hour, respectively, in order to indicate the effect of some of the assumptions of engine weight and airplane characteristics. Separate calculations were made assuming: (1) that in estimating the engine weight, compressor pressure ratios per stage of 1.5 and infinity (compressor weight of zero) are assumed instead of 1.17, which has been assumed in determining the weight per square foot of engine frontal area; (2) that the engines could not be arranged in tandem groups and might therefore require an increase in fuselage diameter in order to submerge the installation; (3) that the engines were placed in nacelles instead of being submerged; and (4) a ratio of structure to gross weight of 0.3 instead of 0.4.

Assumptions necessary for the calculation of (2) and (3) are as follows:

Altitude, ft	30,000	50,000
Flight speed, mph	500	800
Lift-drag ratio of wing.	31.70	14.71

Ratio of tail drag to wing drag	0.25	0.19
Fuselage drag coefficient (based on frontal area) . .	0.089	0.14
Nacelle drag coefficient (based on frontal area). . .	0.074	0.27
Airplane gross weight, lb	100,000	
Fuselage length-diameter ratio.		15
Fuselage volume, cu ft.	$\frac{3\pi^2}{64} d^3 \left(\frac{l}{d}\right)$	
Turbojet engine length, ft.		15
Control volume, cu ft		100
Fuel density, lb/cu ft.		50

The values of the lift-drag and structure-to-gross-weight ratios of the aircraft and the assumed efficiencies of the engine components assumed for the analysis are representative of the best values obtained either in practice or in laboratory investigations.

In order to determine the effect of each engine variable on the load-range performance of the turbojet engine, charts similar to those of reference 1 are used. The dimensionless ratio of disposable load to gross weight W_d/W_g is plotted against the initial fuel consumption per mile per ton of gross weight W_f/W_g . (It should be noted that the units of the gross weight W_g are pounds in all the equations. In the charts where the ratios W_d/W_g and W_f/W_g are plotted against each other in order to compare the performance of various turbojet engines, the gross weight W_g has units of pounds in the ratio W_d/W_g , but has units of tons in the ratio W_f/W_g . All symbols used in this report are defined in the appendix.) On a plot of this type, straight lines through the origin are lines of constant KR, where K is the ratio of the average to the initial fuel rate and R is the range. The relation for KR is

$$KR = \frac{\frac{W_f}{W_g}}{1.05 \frac{W_f}{W_g}} \quad (1)$$

From equation (1), it is easily seen that KR for any point in the chart is equal to the slope of the line joining that point to the origin divided by 1.05 (the ratio of the weight of the fuel and fuel tanks to the weight of the fuel alone). As in reference 1, the value K is calculated by the following equation, which assumes a Breguet type flight plan: (See fig. 5)

$$K = \frac{\frac{1}{1.05} \frac{W_f}{W_g}}{-\log_e \left(1 - \frac{1}{1.05} \frac{W_f}{W_g} \right)} \quad (2)$$

where

$$W_f = W_d - W_c$$

$$W_d = W_g - W_s - W_e$$

$$W_g = F \frac{L}{D} \text{ (for submerged installations)}$$

$$W_g = F \frac{L}{D} \left(1 - \frac{D_n}{F} \right) \text{ (for nacelle installations)}$$

$$W_f' = \frac{fF}{V}$$

The ultimate range is found by setting the pay load W_c equal to zero, for which case

$$\frac{W_f}{W_g} = \frac{W_d}{W_g} = 1 - \frac{W_s}{W_g} - \frac{W_e}{F \frac{L}{D}} \quad (3)$$

or

$$\frac{W_f}{W_g} = \frac{W_d}{W_g} = 1 - \frac{W_s}{W_g} - \frac{W_e}{F \frac{L}{D} \left(1 - \frac{D_n}{F} \right)} \quad (3a)$$

and

$$\frac{W_f'}{W_g} = \frac{f}{V \frac{L}{D}} \quad (4)$$

$$\frac{W_f'}{W_g} = \frac{f}{V} \frac{L}{D} \left(\frac{1}{1 - \frac{D_n}{F}} \right) \quad (4a)$$

where equations (3) and (4) are for the case of a submerged engine installation and equations (3a) and (4a) are for the case of the engine mounted in nacelles.

RESULTS AND DISCUSSION

Engine Performance

The various components of engine performance are discussed in the following paragraphs.

Thrust. - The variation of net thrust per square foot of engine frontal area with compressor pressure ratio for a turbojet engine at altitudes of 10,000, 30,000, 50,000, and 70,000 feet, at flight speeds of 500, 600, 700, and 800 miles per hour, and for turbine-inlet temperatures of 1700°, 2000°, and 2300° R is shown in figure 6.

For the sake of simplicity and convenience, the thrust curves and the curves that follow are not always extended over the entire range of compressor pressure ratios investigated. All the curves of figure 6 show peaking of the thrust with varying compressor pressure ratio, which is characteristic of a turbojet engine with constant turbine-inlet temperature. The maximum thrust varies from about 900 pounds per square foot of engine frontal area at 10,000 feet, 800 miles per hour, and a turbine-inlet temperature of 2300° R (fig. 6(a)) to about 40 pounds per square foot of engine frontal area at 70,000 feet, 500 miles per hour, and a turbine-inlet temperature of 1700° R (fig. 6(d)). The maximum thrust decreases with increasing altitude, and increases with increasing flight speed and turbine-inlet temperature. The compressor pressure ratio for maximum thrust varies with altitude, flight speed, and turbine-inlet temperature, but pressure ratios from 5 to 10 give maximum or near maximum values for the thrust at all conditions investigated.

Specific weight. - The variation of specific engine weight with compressor pressure ratio is plotted in figure 7 for the same range of conditions given for figure 6. The specific engine weight increases with increasing compressor pressure ratio throughout the range of

pressure ratios investigated. There is no minimum with varying compressor pressure ratio because the engine weight decreases more rapidly than the thrust as the pressure ratio decreases from the value necessary for maximum thrust. Specific weight increases with increasing altitude, and decreases with increasing turbine-inlet temperature. Increasing the flight speed causes a decrease in specific weight at low pressure ratios, but causes an increase in specific weight at high pressure ratios. The pressure ratio at which the change in the effect of flight speed occurs increases with increasing turbine-inlet temperature.

Specific fuel consumption. - The variation of thrust specific fuel consumption with compressor pressure ratio is shown in figure 8 for the same range of conditions given for figures 6 and 7. The specific-fuel-consumption curves have a minimum point with respect to varying pressure ratio, which again is characteristic of a turbojet engine with a constant turbine-inlet temperature. The minimum specific fuel consumption decreases with increasing altitude and increases with increasing flight speed, varying from about 0.80 at 70,000 feet and 500 miles per hour (fig. 8(d)) to about 1.5 at 10,000 feet and 800 miles per hour (fig. 8(a)). The minimum specific fuel consumption varies only slightly with the turbine-inlet temperature in the range of temperatures investigated. The compressor pressure ratio at which the minimum specific fuel consumption occurs decreases with increasing flight speed and increases with increasing altitude and turbine-inlet temperature, varying from about 7.5 at 10,000 feet, 800 miles per hour, and a turbine-inlet temperature of 1700° R, to values greater than 30 (above the range of pressure ratios investigated) at all flight speeds at 50,000 and 70,000 feet with a turbine-inlet temperature of 2300° R.

Load-Range Characteristics

In order to compare the load-range performance of the various turbojet engines, charts of the type described in the section METHODS are used. The load-range performance of the turbojet engine is shown in figure 9 for the same conditions given for the engine-performance curves. (The values of W_d/W_g and W_F/W_g used to plot fig. 9 are listed in table I.)

If the flight speed and lift-drag ratio are constant, as they are in each individual plot of figure 9, the ratio W_d/W_g depends only upon the specific engine weight and the ratio W_F/W_g depends only upon the specific fuel consumption. (W_d/W_g decreases as the

specific engine weight increases, and W_f/W_g increases with specific fuel consumption.) The variation of these ratios with the compressor pressure ratio and the turbine-inlet temperature over the range of conditions investigated therefore follows directly from the variation of the specific weight and the specific fuel consumption. (See figs. 7 and 8, respectively.) The ultimate range and the range with pay load are determined by the values of the ratios W_d/W_g and W_f/W_g , as described in the section METHODS.

For any point on the curves of figure 9, KR is equal to the slope of the line joining that point to the origin of the coordinate system (0,0) divided by 1.05 (the ratio of the weight of the fuel and fuel tanks to the weight of the fuel alone). Three lines of constant KR are shown on each plot of figure 9 for convenience in estimating range.

Ultimate range. - In order to show more conveniently the effect of the different variables on the ultimate range, the following tabulation of the maximum ultimate range and the compressor pressure ratio at which this maximum occurs for all the flight conditions and turbine-inlet temperatures investigated, using figures 9 and 5, is presented.

Altitude (ft)	Turbine- inlet tempera- ture (°R)	Flight speed (mph)							
		500		600		700		800	
		Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c
10,000	1700	6550	10	5900	8	4100	6	3400	5
	2000	6450	15	6150	12	4150	10	3500	8
	2300	6400	25	6000	19	4100	14	3450	12
30,000	1700	7800	12	7600	10	5300	9	4850	8
	2000	7700	18	7650	15	5350	13	5000	12
	2300	7400	25	7500	20	5300	16	5050	15
50,000	1700	6200	9	5000	8	3950	7	4300	6
	2000	6250	13	5050	10	4000	8	4500	7
	2300	6100	16	5150	13	4100	9	4700	8
70,000	1700	3600	6.5	2600	4.5	1850	3	2300	2.5
	2000	4000	7	3000	5	2200	4	2550	3.5
	2300	4200	7.5	3300	5.5	2400	5	2800	4

The maximum ultimate range increases as the altitude is increased to about 30,000 feet, and then decreases as the altitude is further increased. Due to the decrease in lift-drag ratio, the ultimate range decreases with increasing flight speed in the transonic region except where the lift-drag ratio remains nearly constant (700 to 800 mph at 50,000 and 70,000 ft); here the ultimate range increases with flight speed. For all speeds at altitudes of 10,000 and 30,000 feet, and for speeds up to 700 miles per hour at 50,000 feet, the variation of ultimate range with turbine-inlet temperature, within the range investigated, is about 5 percent or less, with a temperature of 1700°R giving the longest range at the lower speeds and altitudes. At 50,000 feet and 800 miles per hour, and at all speeds at 70,000 feet, a turbine-inlet temperature of 2300°R gives a 10 to 30 percent longer range than the temperature of 1700°R , and a 4 to 10 percent longer range than the temperature of 2000°R .

The compressor pressure ratio for maximum ultimate range increases as the altitude is increased to 30,000 feet and then decreases as the altitude continues to increase. At all altitudes, the pressure ratio for maximum ultimate range decreases with increasing flight speed and increases with increasing turbine-inlet temperature.

The maximum ultimate range found at any of the conditions investigated is 7800 miles at 30,000 feet and 500 miles per hour, with a turbine-inlet temperature of 1700°R , a lift-drag ratio of 20, and a compressor pressure ratio of 12. At a flight speed of 800 miles per hour, the longest ultimate range found at any of the conditions investigated is 5050 miles at 30,000 feet with a turbine-inlet temperature of 2300°R , a lift-drag ratio of 10, and a compressor pressure ratio of 15.

As may be seen from figure 9, the ultimate range falls off slowly as the pressure ratio is varied in either direction from the pressure ratio necessary for maximum ultimate range; that is, there exists at each flight condition a range of compressor pressure ratios that will give close to optimum performance on the basis of ultimate range. This range of pressure ratios is wider at low flight speeds than at high flight speeds with the same variation in performance. Some latitude in the selection of design compressor pressure ratio for a given application exists because of this band of pressure ratios giving close to optimum ultimate-range performance. A compressor pressure ratio of about 8 to 10 will give optimum, or close to optimum, ultimate range at all the conditions investigated.

Ranges less than ultimate. - Another measure of the load-range performance of the turbojet engine is the range with a given pay load. Figure 10 shows, for a turbojet operating at 50,000 feet and 800 miles per hour with turbine-inlet temperatures of 1700°, 2000°, and 2300° R, the variation of range with compressor pressure ratio for values of the pay-load- to-gross-weight ratio W_c/W_g of 0.0 (ultimate range), 0.2, and 0.4. Tabulations of the maximum range, and the compressor pressure ratio at which the maximum range occurs, for the flight conditions and pay loads of figure 10, and for all flight conditions for a pay-load- to-gross-weight ratio W_c/W_g of 0.2 follow:

W_c/W_g	Turbine-inlet temperature (°R)	Maximum range (miles)	r_c
0.0	1700	4300	6
	2000	4500	7
	2300	4700	8
0.2	1700	2150	3.5
	2000	2300	4.5
	2300	2400	7
0.4	1700	640	<2
	2000	675	3.5
	2300	725	4

$w_c/w_g = 0.2$									
Altitude (ft)	Turbine- inlet tempera- ture (°R)	Flight speed, (mph)							
		500		600		700		800	
		Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c	Maximum ultimate range (miles)	r_c
10,000	1700	3750	8.5	3300	7.5	2300	5.5	1900	4.5
	2000	3600	13.5	3400	11	2300	9	1900	7
	2300	3600	21	3400	16	2300	13	1900	10.5
30,000	1700	4250	10	4200	9	2850	7.5	2650	6.5
	2000	4150	15	4200	12.5	2900	10	2700	8.5
	2300	4050	22	4150	18	2900	14.5	2750	11.5
50,000	1700	3200	6.5	2500	5.5	1950	4.5	2150	3.5
	2000	3200	8.5	2600	7	2050	5.5	2300	4.5
	2300	3200	11.5	2650	9	2100	8	2400	7
70,000	1700	1350	4	950	2	650	<2	850	<2
	2000	1700	5	1200	3	850	2	1100	<2
	2300	1950	6	1400	4.5	950	3	1200	2

For the flight conditions of figure 10, the maximum range and the corresponding compressor pressure ratio both decrease as the pay load increases. Although it is not shown, this decrease also occurs for all other flight conditions. The maximum range and the compressor pressure ratio at which it occurs follow the same trends with altitude and flight speed as for the case of maximum ultimate range.

The trends with turbine-inlet temperature for this case are similar to those of the maximum ultimate range, as previously discussed, inasmuch as an increase in turbine-inlet temperature results in an increase in the maximum range and corresponding compressor pressure ratio for high flight speeds and altitudes (as illustrated in fig. 10 for one set of flight conditions). An increase in turbine-inlet temperature results, however, in a decrease in the maximum range and an increase in the corresponding pressure ratio for low flight speeds and altitudes within the range of flight conditions investigated. At values of the pay load approaching the total disposable load, however, the maximum range increases with increasing turbine-inlet temperature at all flight conditions.

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As in the case of the ultimate range, there exists for each pay load at each flight condition a range of compressor pressure ratios that will give close to the maximum range (fig. 10). For a value of pay-load- to-gross-weight ratio of 0.2, a compressor pressure ratio of about 5 to 7 will give optimum, or close to optimum, range at all of the flight conditions investigated except at an altitude of 70,000 feet and flight speeds of 700 and 800 miles per hour, where a lower pressure ratio is required.

Ultimate range at subsonic and supersonic flight speeds. - Some indication of how the load-range performance of the turbojet engine in the region of transonic flight speeds compares with the load-range performance of the turbojet engine at lower and higher flight speeds is given in figure 11. The variation of maximum ultimate range with flight speed is shown for flight speeds from 0 to 1400 miles per hour at an altitude of 50,000 feet with the combination of turbine-inlet temperature and compressor pressure ratio that produces the maximum range at each flight condition. The ultimate range increases as the flight speed increases until the transonic region is reached (500 to 550 mph). Here the sudden decrease in lift-drag ratio (fig. 3) results in a marked decrease in range. When the transonic region is passed (at about 700 mph) and the lift-drag ratio becomes fairly constant again, increasing the flight speed tends to increase the range. The range does not increase as rapidly as in the subsonic region because the lift-drag ratio is decreasing slowly in the supersonic region, rather than being constant as is the case in the subsonic region. The range therefore increases slowly as the flight speed increases, until the wing loading, which is also increasing with increasing flight speed, reaches the maximum permissible value (at about 1100 mph). As the design flight speed further increases, the wing-loading limitation causes the flight lift-drag ratio to decrease and the maximum ultimate range rapidly falls off. If the altitude is such that the wing-loading limitation is reached at some flight speed in the subsonic or transonic region, the ultimate range decreases thereafter and there is no range increase in the supersonic region.

Effect of Changes in Engine Weight and Airplane Characteristics

The effects on load-range performance of a change in some of the assumptions for engine weight and airplane characteristics are discussed in the following sections. One assumption is changed in each section. The flight conditions considered are 30,000 feet, 500 miles per hour and 50,000 feet, 800 miles per hour.

Compressor weight. - The compressor weight, and consequently the engine weight, depends upon the compressor pressure ratio per stage, which was assumed equal to 1.17 in the previous calculations. The following table shows the effect of assuming pressure ratios per stage of 1.5 and infinity (compressor weight of zero) instead of the 1.17 previously assumed:

Flight conditions	Turbine-inlet temperature (°R)	$r_c' = 1.17$		$r_c' = 1.5$		$r_c' = \infty$	
		Ultimate range (miles)	r_c	Ultimate range (miles)	r_c	Ultimate range (miles)	r_c
30,000 ft, 500 mph	1700	7800	12	8050	13	8300	13.5
	2000	7700	18	7900	19.5	8150	20.5
	2300	7400	25	7650	27	7850	28
50,000 ft, 800 mph	1700	4300	6	4650	6.5	5000	7
	2000	4500	7	4800	8	5100	8.5
	2300	4700	8	5000	9.5	5300	10.5

Changing the pressure ratio per stage to 1.5 and infinity resulted in increases in ultimate range of about 3 and 6 percent, respectively, at 30,000 feet and 500 miles per hour, and about 7 and 15 percent, respectively, at 50,000 feet and 800 miles per hour. The compressor pressure ratio for maximum range increased as the pressure ratio per stage increased; the variation of ultimate range with turbine-inlet temperature remained about the same.

Engine installation. - In the previous calculations, the engines were assumed to be submerged in the fuselage. When the airplane lift-drag ratios assumed in METHODS are checked by use of the assumed wing lift-drag ratios and companion assumptions listed in METHODS, it appears possible to submerge the engines without increasing the fuselage volume above that corresponding to the assumed lift-drag ratio. At some flight conditions, however, it is necessary to arrange the engines in tandem groups in order not to increase the fuselage diameter. This arrangement is referred to in the following table as "minimum fuselage diameter." If this arrangement is undesirable and it is necessary to keep the engines in one group, it would be necessary, for some flight conditions, to enlarge the fuselage diameter to accommodate the engines. Also, it is possible that a submerged installation is undesirable and that the engines must be mounted in nacelles.

The following table shows the variation of the maximum ultimate range and the corresponding compressor pressure ratio with these three different engine installations:

Flight conditions	Turbine-inlet temperature ($^{\circ}$ R)	Minimum fuselage diameter		Enlarged fuselage diameter		Engine in nacelles	
		Ultimate range (miles)	r_c	Ultimate range (miles)	r_c	Ultimate range (miles)	r_c
30,000 ft, 500 mph	1700	7800	12	7800	12	7050	11
	2000	7700	18	7700	18	7000	17
	2300	7400	25	7400	25	6950	24
50,000 ft, 800 mph	1700	4300	6	4050	6	1300	5.5
	2000	4500	7	4300	7	2150	6.5
	2300	4700	8	4600	8	2750	7.5

At 30,000 feet and 500 miles per hour, it is possible to submerge the engines in the fuselage without enlarging the fuselage diameter, even if the engines must be arranged in one single-plane group. The minimum and enlarged fuselage diameters are therefore equal for this flight condition and the load-range performances of these two installations are identical. At 50,000 feet and 800 miles per hour, enlargement of the fuselage diameter to accommodate the engines resulted in a decrease in range of from about 2 to 6 percent. When the engines were mounted in nacelles instead of being completely submerged, a decrease in ultimate range resulted of about 5 to 10 percent at 30,000 feet and 500 miles per hour, and about 40 to 70 percent at 50,000 feet and 800 miles per hour. At 30,000 feet and 500 miles per hour, a turbine-inlet temperature of 1700° R gives the longest ultimate range regardless of the engine installation. At 50,000 feet and 800 miles per hour, a turbine-inlet temperature of 2300° R gives the longest ultimate range regardless of the engine installation, and the margin of superiority at this temperature becomes very large in the case of a nacelle installation. The compressor pressure ratio for maximum range remained about the same for the submerged installations and decreased slightly for the nacelle installation. The table indicates the extreme desirability of submerged engine installations for transonic and supersonic flight speeds.

Structure weight. - The ratio of structure to gross weight W_s/W_g was assumed equal to 0.4 for the previous calculations. The following table shows the effect of assuming a structure-to-gross-weight ratio of 0.3 instead of 0.4:

Flight conditions	Turbine-inlet temperature (°R)	$W_s/W_g = 0.4$		$W_s/W_g = 0.3$	
		Ultimate range (miles)	r_c	Ultimate range (miles)	r_c
30,000 ft, 500 mph	1700	7800	12	9900	13
	2000	7700	18	9800	19
	2300	7400	25	9500	26
50,000 ft, 800 mph	1700	4300	6	5700	7
	2000	4500	7	5900	8
	2300	4700	8	6100	9

Changing the structure-to-gross-weight ratio from 0.4 to 0.3 resulted in an increase in ultimate range of about 30 percent; the corresponding compressor pressure ratio was increased and the variation of ultimate range with turbine-inlet temperature remained the same.

SUMMARY OF RESULTS

The results of an analysis of the engine performance and load-range characteristics of the turbojet engine for flight speeds of 500, 600, 700, and 800 miles per hour, flight altitudes of 10,000, 30,000, 50,000, and 70,000 feet, turbine-inlet temperatures of 1700°, 2000°, and 2300° R, and compressor pressure ratios from 2 to 30 may be summarized as follows:

1. The operating altitude that gave the longest range at the flight speeds investigated was about 30,000 feet.
2. The maximum ultimate range at the conditions investigated occurred at a flight speed of 500 miles per hour. The ultimate ranges decreased rapidly as the speed was increased in the transonic region (550 to 700 mph). If the wing-loading limit had not been reached, the ultimate range increased slightly with increasing flight speed in the supersonic region (above 700 mph) until the wing-loading limit was reached and then the range rapidly fell off.
3. With the engines submerged, the variation of ultimate range with turbine-inlet temperature for the temperature range investigated was about 5 percent or less for all flight speeds at 10,000 and 30,000 feet and for speeds up to 700 miles per hour at 50,000 feet, with a turbine-inlet temperature of 1700° R giving the longest

range at the lower speeds at an altitude of 10,000 feet. At 50,000 feet and 800 miles per hour and at all speeds at 70,000 feet, a turbine-inlet temperature of 2300°R gave a 10 to 30 percent longer range than the temperature of 1700°R and a 4 to 10 percent longer range than the temperature of 2000°R .

4. The compressor pressure ratio for maximum ultimate range varied greatly, increasing as the altitude increased to about 30,000 feet, and then decreasing; increasing with turbine-inlet temperature; and decreasing with increasing flight speed. At any flight condition within the range investigated, however, optimum or near optimum ultimate range can be attained with a compressor pressure ratio of about 8 to 10.

5. The maximum ultimate range attainable by the turbojet engine within the range of conditions investigated was 7800 miles at 30,000 feet and 500 miles per hour with a turbine-inlet temperature of 1700°R , a lift-drag ratio of 20, and a compressor pressure ratio of 12. At a flight speed of 800 miles per hour, the longest ultimate range found at any of the conditions investigated was 5050 miles at 30,000 feet with a lift-drag ratio of 10, as turbine-inlet temperature of 2300°R , and a compressor pressure ratio of 15. The ratio of structure to gross weight was 0.4 for both these cases.

6. The range of an aircraft carrying a given pay load followed the same general trends with varying flight conditions, turbine-inlet temperature, and compressor pressure ratio as did the ultimate range. As the pay load increased, however, lower pressure ratios and higher turbine-inlet temperatures gave maximum range. At a pay-load-to-gross-weight ratio of 0.2, optimum, or close to optimum, range can be obtained at most flight conditions investigated with a compressor pressure ratio of about 5 to 7.

7. The assumption of compressor pressure ratios per stage of 1.5 and infinity, instead of 1.17, resulted in an increase in range of about 3 and 6 percent, respectively, at 30,000 feet and 500 miles per hour, and about 7 and 15 percent, respectively, at 50,000 feet and 800 miles per hour. The trends of the ultimate range with turbine-inlet temperature and compressor pressure ratio remained about the same.

8. If it were required that the engines be arranged in a single-plane group, as opposed to tandem grouping, it would be necessary at some flight conditions to enlarge the fuselage diameter in order to submerge them. At 50,000 feet and 800 miles per hour, the enlargement of the fuselage diameter would result in a decrease in ultimate range

of from about 2 to 6 percent. At 30,000 feet and 500 miles per hour, it would be unnecessary to enlarge the fuselage diameter. If the engines were mounted in nacelles rather than submerged, a decrease in ultimate range of about 5 to 10 percent at 30,000 feet and 500 miles per hour and about 40 to 70 percent at 50,000 feet and 800 miles per hour would result. The longest ultimate range, regardless of the engine installation, would occur at a turbine-inlet temperature of 1700° R at 30,000 feet and 500 miles per hour, and at a turbine-inlet temperature of 2300° R at 50,000 feet and 800 miles per hour for the temperature range studied. The variation of ultimate range with compressor pressure ratio remained about the same, regardless of engine installation.

9. The assumption that the structure- to-gross-weight ratio was 0.3 instead of 0.4 resulted in an increase in ultimate range of about 30 percent at 30,000 feet, 500 miles per hour and at 50,000 feet, 800 miles per hour. The trends of the ultimate range with turbine-inlet temperature and compressor pressure ratio were unchanged.

Lewis Flight Propulsion Laboratory,
National Advisory Committee for Aeronautics,
Cleveland, Ohio, November 29, 1949.

APPENDIX - SYMBOLS

The following symbols are used throughout this report:

D_n	nacelle drag per unit engine frontal area, lb/sq ft
d	fuselage diameter, ft
F	net thrust per unit engine frontal area, lb/sq ft
f	thrust specific fuel consumption, lb/lb/hr
K	ratio of average fuel rate to initial fuel rate
L/D	lift-drag ratio of aircraft without nacelles
l	fuselage length, ft
R	range, miles
r_c	compressor pressure ratio
r_c'	compressor pressure ratio per stage
V	flight speed, mph
W_c	pay load per unit engine frontal area, lb/sq ft
W_d	disposable load per unit engine frontal area, lb/sq ft
W_e	engine weight per unit engine frontal area, lb/sq ft
W_f	fuel plus fuel-tank weight per unit engine frontal area, lb/sq ft
W_f'	initial fuel rate per unit engine frontal area, lb/mile/sq ft
W_g	gross weight per unit engine frontal area, lb/sq ft
W_s	structure weight per unit engine frontal area, lb/sq ft

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2. Keenan, Joseph H., and Kaye, Joseph: Thermodynamic Properties of Air. John Wiley & Sons, Inc., 1945.
3. Pinkel, Benjamin; and Karp, Irving M.: A Thermodynamic Study of the Turbojet Engine. NACA Rep. 891, 1947.
4. Pinkel, Benjamin, and Turner, L. Richard: Thermodynamic Data for the Computation of the Performance of Exhaust-Gas Turbines. NACA ARR 4B25, 1944.

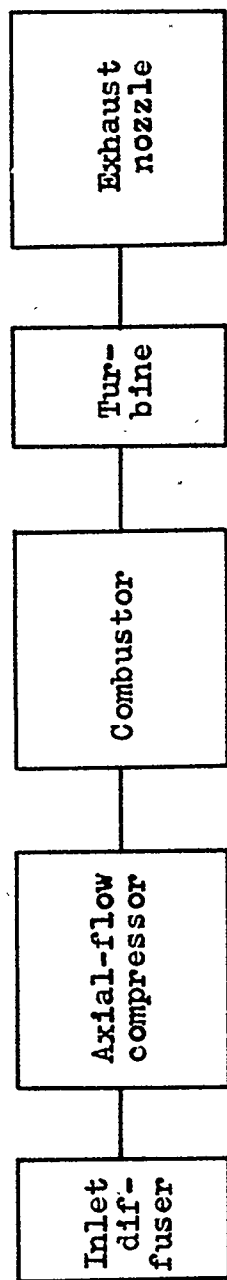
TABLE I - LOAD RANGE FACTORS W_d/W_g

Turbine-inlet temperature (°R)	Compressor pressure ratio	Flight speed (mph)							
		500		600		700		800	
		W _d /W _g	W _f '/W _g	W _d /W _g	W _f '/W _g	W _d /W _g	W _f '/W _g	W _d /W _g	W _f '/W _g
		Altitude, 10,000 feet							
1700	2	0.579	0.378	0.577	0.385	0.566	0.520	0.561	0.572
	5	.571	.264	.567	.274	.549	.364	.536	.413
	10	.554	.225	.540	.243	.504	.346	.463	.427
	20	.254	.419	-----	-----	-----	-----	-----	-----
	30	-----	-----	-----	-----	-----	-----	-----	-----
2000	2	.583	.416	.581	.416	.574	.546	.569	.618
	5	.578	.284	.575	.290	.563	.396	.555	.440
	10	.568	.241	.561	.249	.541	.343	.525	.398
	20	.534	.225	.512	.249	.452	.358	.372	.473
	30	.451	.275	-----	-----	-----	-----	-----	-----
2300	2	.585	.449	.584	.449	.577	.588	.575	.627
	5	.582	.309	.579	.314	.570	.421	.563	.480
	10	.575	.259	.570	.266	.555	.368	.545	.419
	20	.555	.231	.546	.240	.515	.334	.489	.392
	30	.532	.218	.511	.240	.454	.339	.400	.403
		Altitude, 30,000 feet							
1700	2	0.570	0.315	0.568	0.304	0.558	0.377	0.554	0.381
	5	.559	.216	.556	.213	.535	.278	.530	.289
	10	.539	.184	.532	.182	.497	.242	.483	.257
	20	.467	.180	.427	.194	.282	.288	.120	.364
	30	.246	.246	-----	-----	-----	-----	-----	-----
2000	2	.575	.342	.574	.322	.566	.405	.563	.405
	5	.566	.243	.565	.233	.550	.300	.547	.307
	10	.553	.202	.550	.196	.527	.259	.520	.270
	20	.521	.178	.507	.179	.458	.237	.430	.256
	30	.468	.180	.442	.182	.341	.255	.242	.292
2300	2	.578	.372	.577	.349	.571	.437	.568	.437
	5	.572	.258	.571	.251	.558	.321	.557	.330
	10	.562	.218	.559	.213	.542	.275	.538	.286
	20	.539	.192	.533	.185	.498	.249	.486	.261
	30	.512	.181	.499	.181	.451	.239	.410	.273

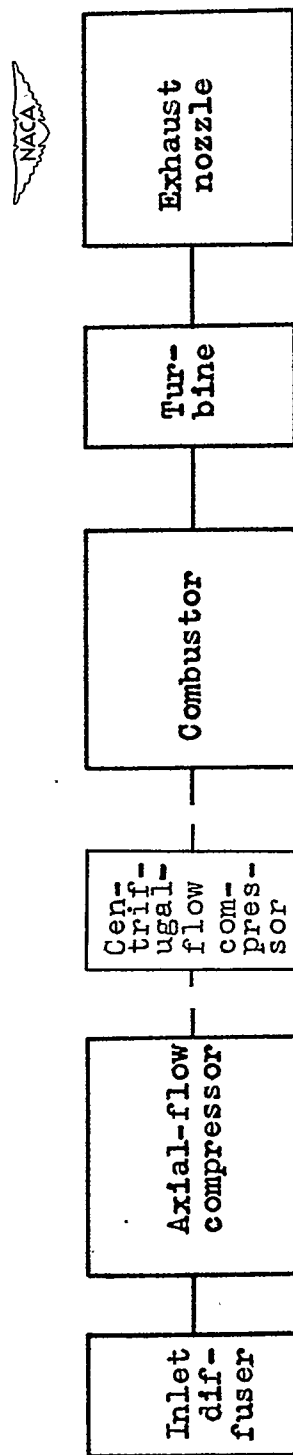


AND W_f'/W_g (USED IN PLOTTING FIGURE 9)

Turbine-inlet temperature (°R)	Compressor pressure ratio	Flight speed (mph)							
		500		600		700		800	
		W _d /W _g	W _f '/W _g	W _d /W _g	W _f '/W _g	W _d /W _g	W _f '/W _g	W _d /W _g	W _f '/W _g
		Altitude, 50,000 feet							
1700	2	0.521	0.312	0.505	0.338	0.488	0.384	0.503	0.322
	5	.507	.217	.477	.244	.450	.283	.452	.260
	10	.456	.183	.415	.207	.363	.247	.359	.230
	20	.320	.169	.204	.200	-----	-----	-----	-----
	30	-----	-----	-----	-----	-----	-----	-----	-----
2000	2	.533	.343	.521	.367	.509	.411	.518	.367
	5	.524	.238	.502	.265	.482	.306	.486	.280
	10	.489	.200	.461	.225	.428	.263	.428	.246
	20	.416	.179	.357	.202	.277	.241	.266	.228
	30	.316	.171	.209	.198	.046	.246	-----	-----
2300	2	.543	.363	.532	.397	.521	.444	.531	.392
	5	.535	.257	.518	.285	.502	.330	.506	.298
	10	.508	.218	.487	.243	.462	.283	.470	.253
	20	.458	.192	.417	.215	.365	.254	.370	.234
	30	.396	.182	.334	.204	.252	.243	.240	.232
		Altitude, 70,000 feet							
1700	2	0.392	0.314	0.358	0.335	0.313	0.382	0.339	0.342
	5	.343	.218	.287	.242	.198	.292	.214	.260
	10	.248	.177	.128	.205	-----	-----	-----	-----
	20	-----	-----	-----	-----	-----	-----	-----	-----
	30	-----	-----	-----	-----	-----	-----	-----	-----
2000	2	.428	.337	.399	.363	.366	.409	.389	.364
	5	.393	.238	.348	.264	.297	.302	.302	.279
	10	.321	.200	.242	.225	.131	.277	.157	.245
	20	.121	.179	-----	-----	-----	-----	-----	-----
	30	-----	-----	-----	-----	-----	-----	-----	-----
2300	2	.451	.361	.425	.393	.398	.439	.421	.388
	5	.428	.252	.390	.281	.346	.328	.357	.295
	10	.367	.219	.309	.242	.244	.282	.249	.262
	20	.230	.192	.129	.215	-----	-----	-----	-----
	30	.069	.182	-----	-----	-----	-----	-----	-----



(a) Compressor pressure ratios less than or equal to 10.



(b) Compressor pressure ratios greater than 10.

Figure 1. - Diagram of assumed turbojet engine.

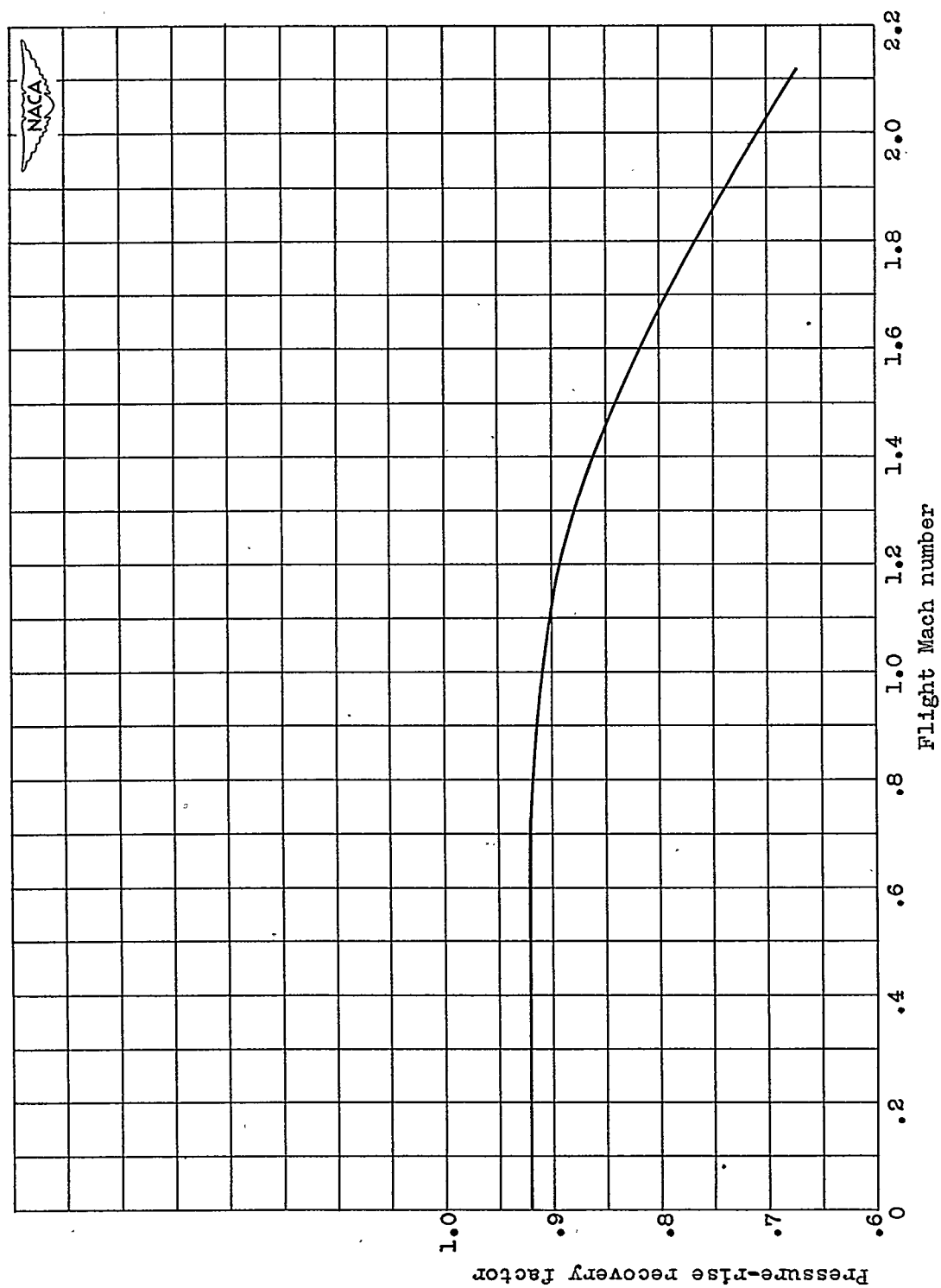


Figure 2. - Variation of recovery factor of engine-inlet-diffuser pressure rise with flight Mach number.

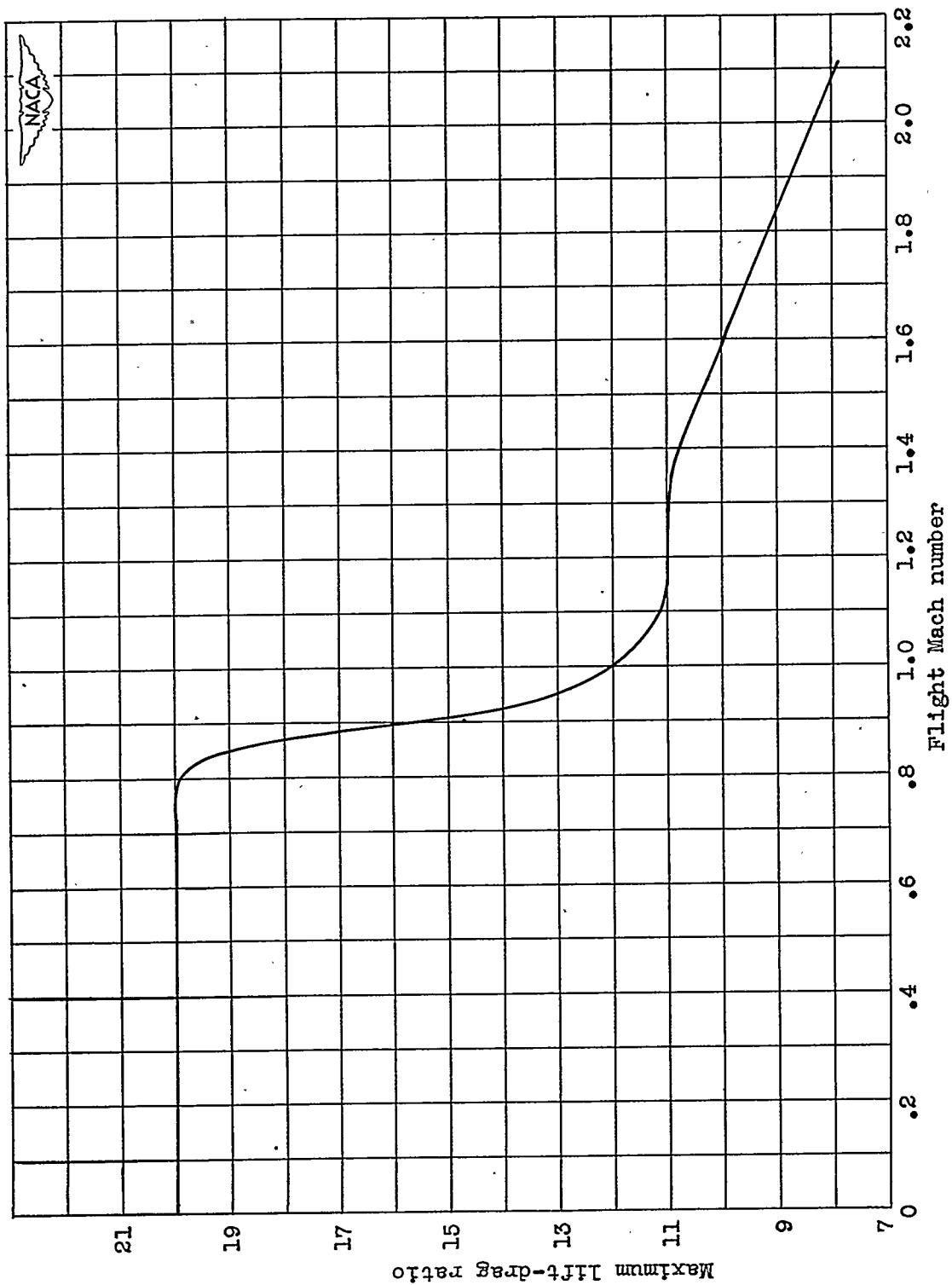


Figure 3. - Variation of maximum airplane lift-drag ratio with flight Mach number.

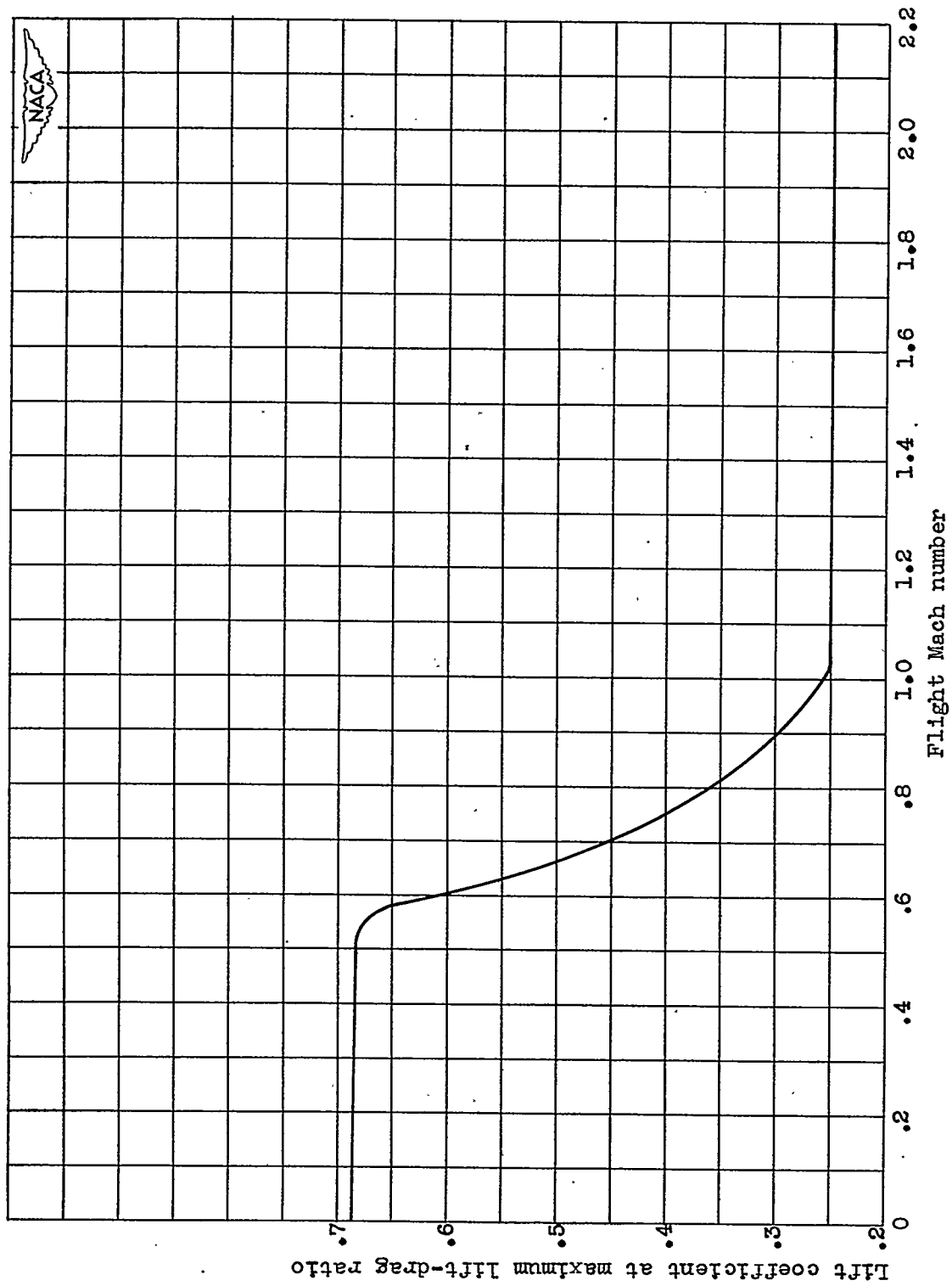


Figure 4. - Variation of lift coefficient at maximum lift-drag ratio with flight Mach number.

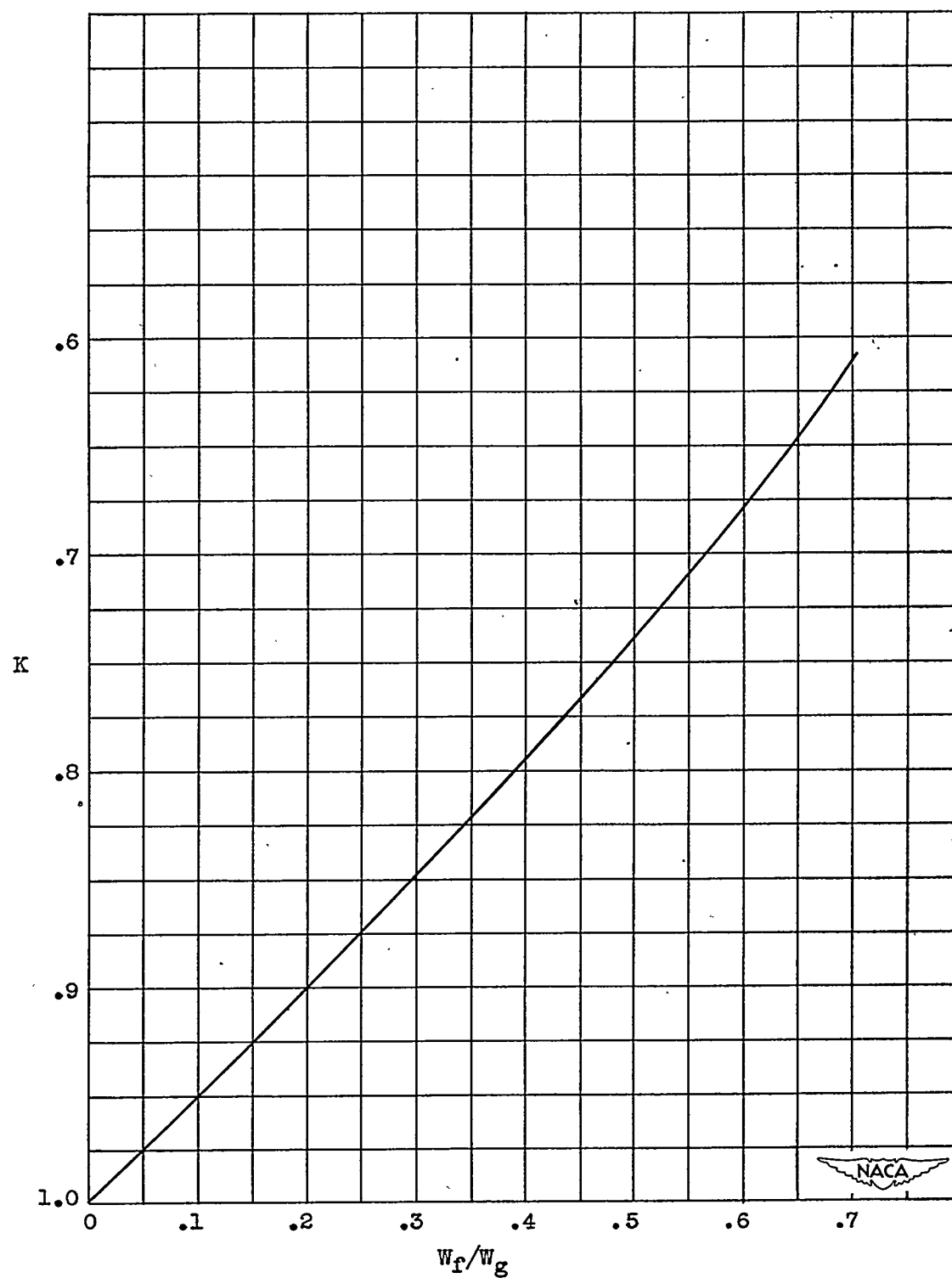
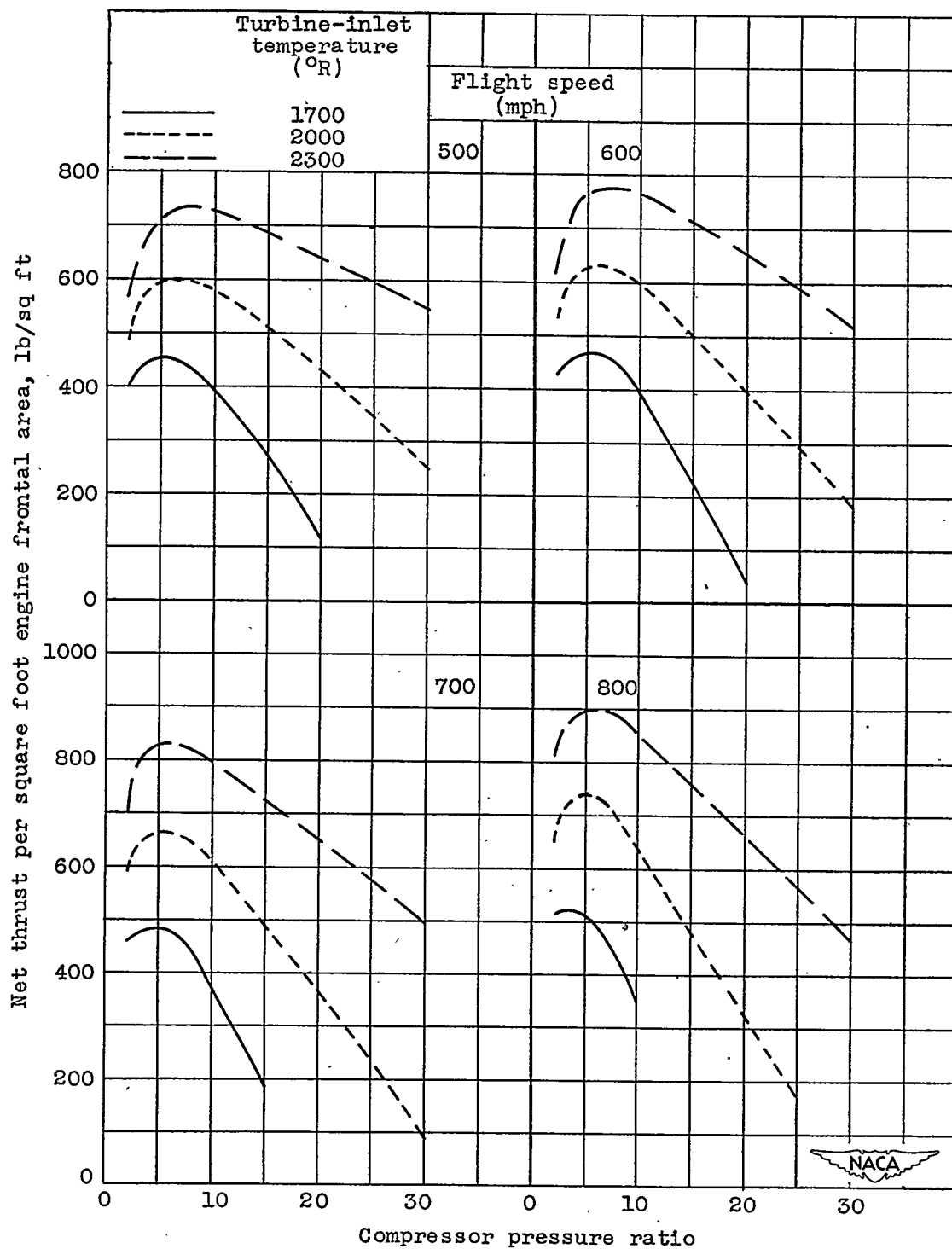
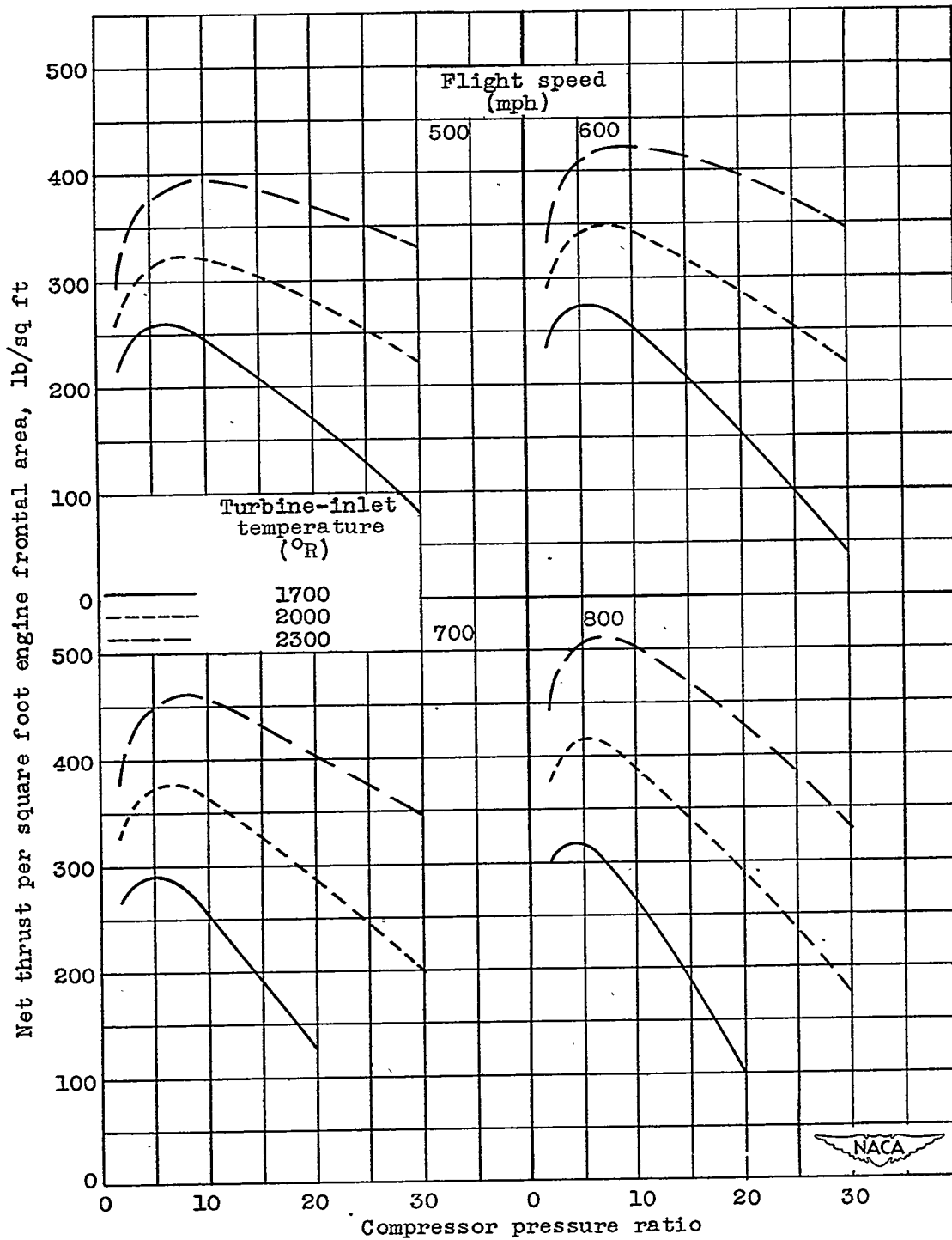


Figure 5. - Variation of K with W_f/W_g .



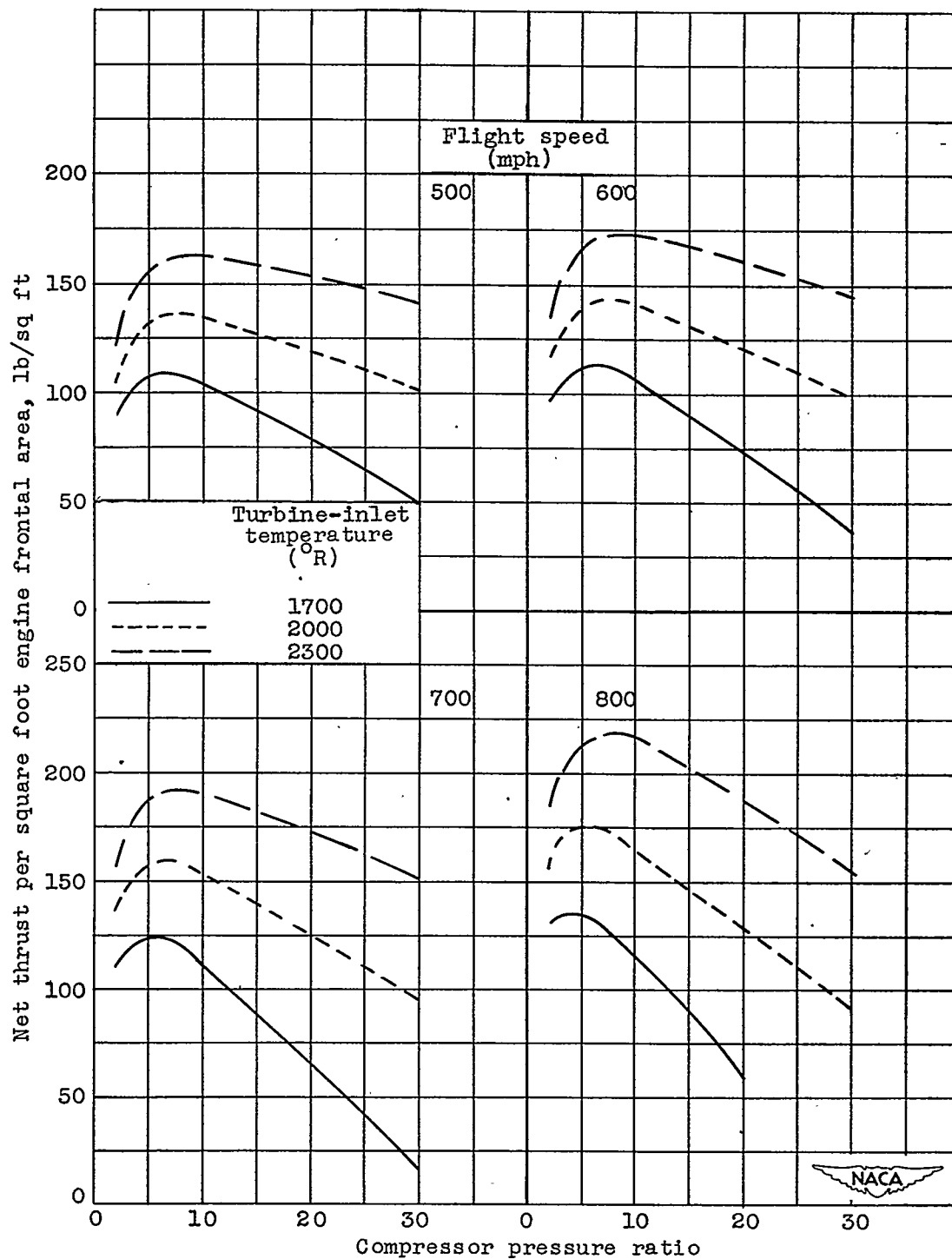
(a) Altitude, 10,000 feet.

Figure 6. - Variation of net thrust per square foot engine frontal area with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



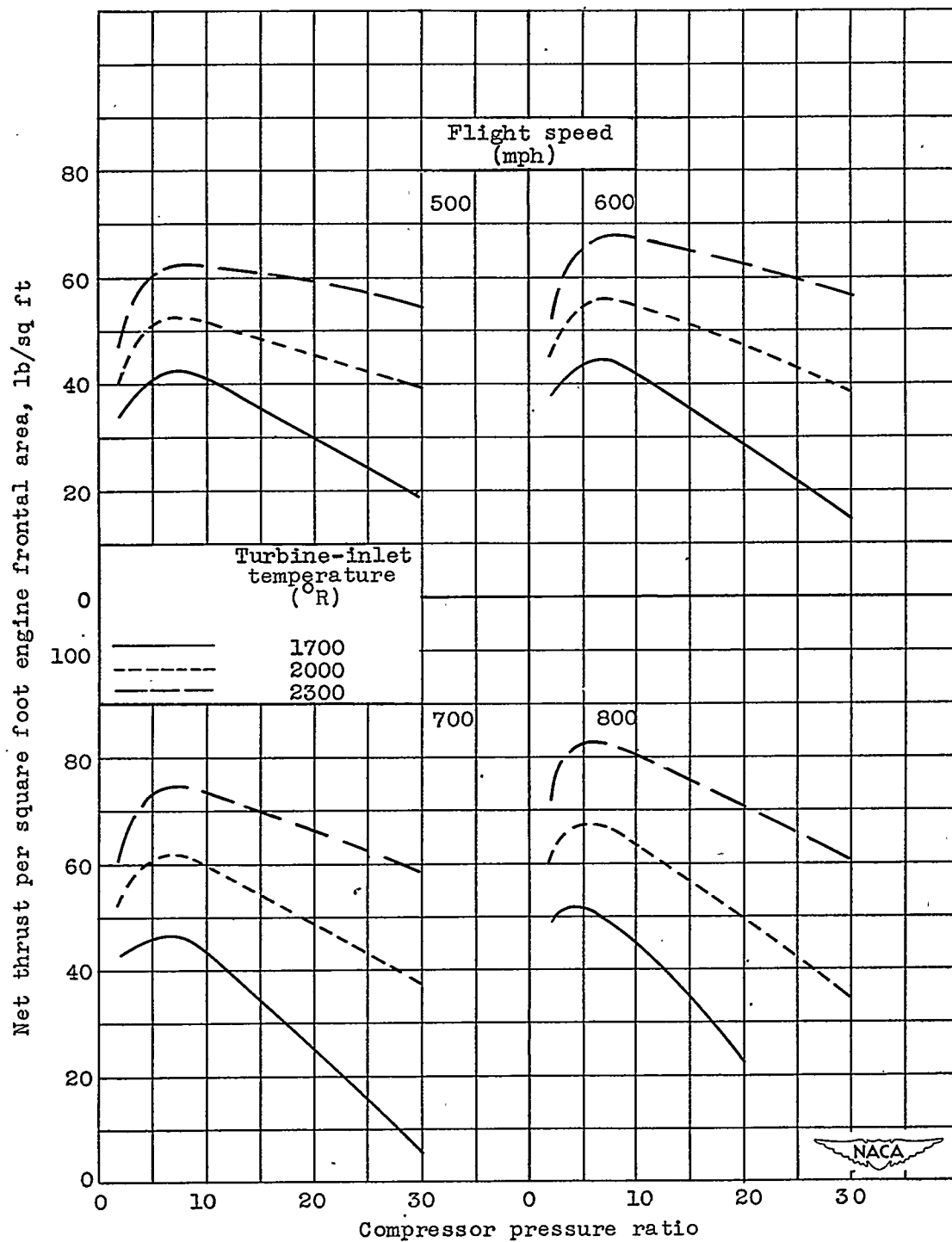
(b) Altitude, 30,000 feet.

Figure 6. - Continued. Variation of net thrust per square foot engine frontal area with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



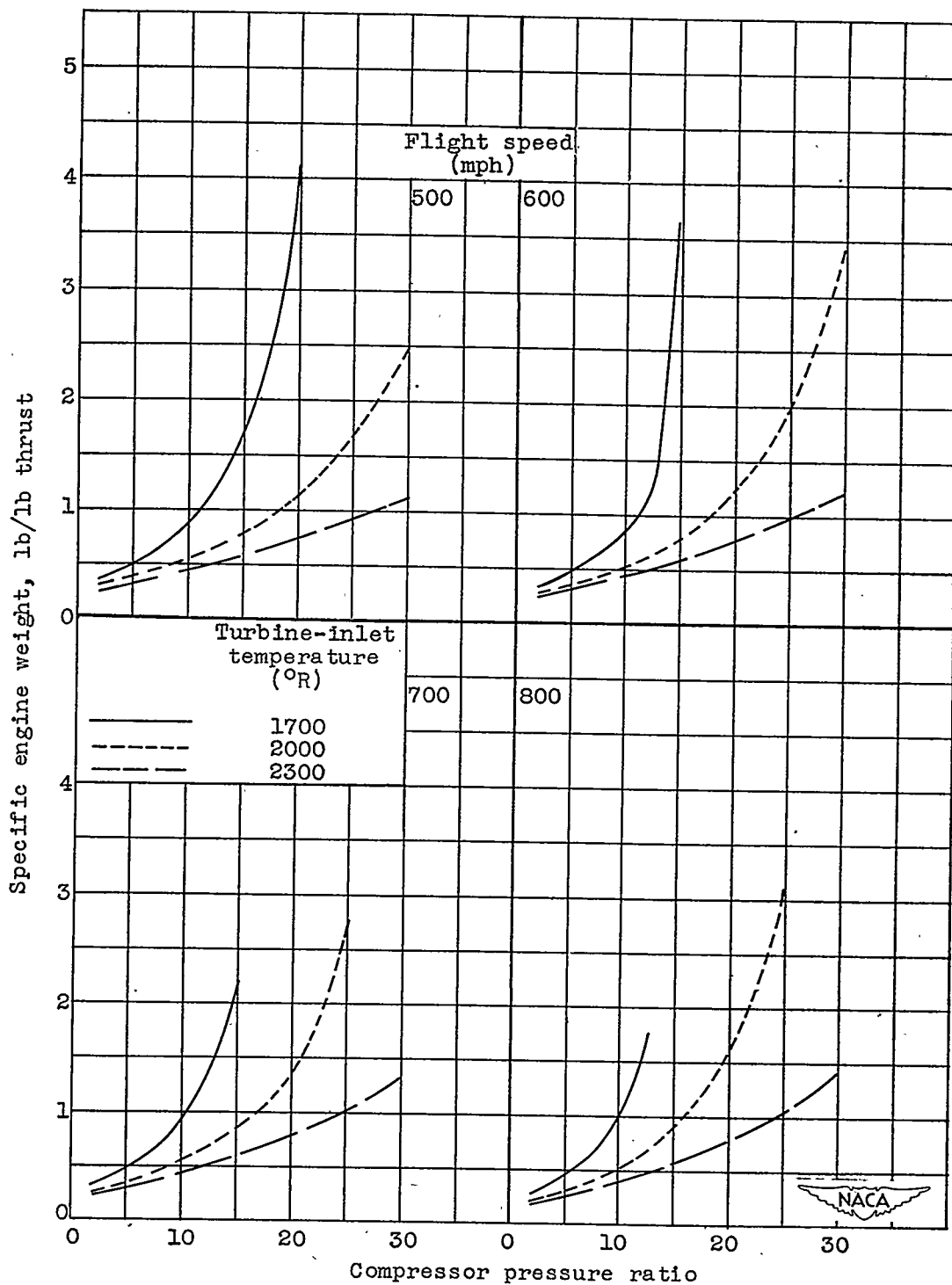
(c) Altitude, 50,000 feet.

Figure 6. - Continued. Variation of net thrust per square foot engine frontal area with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



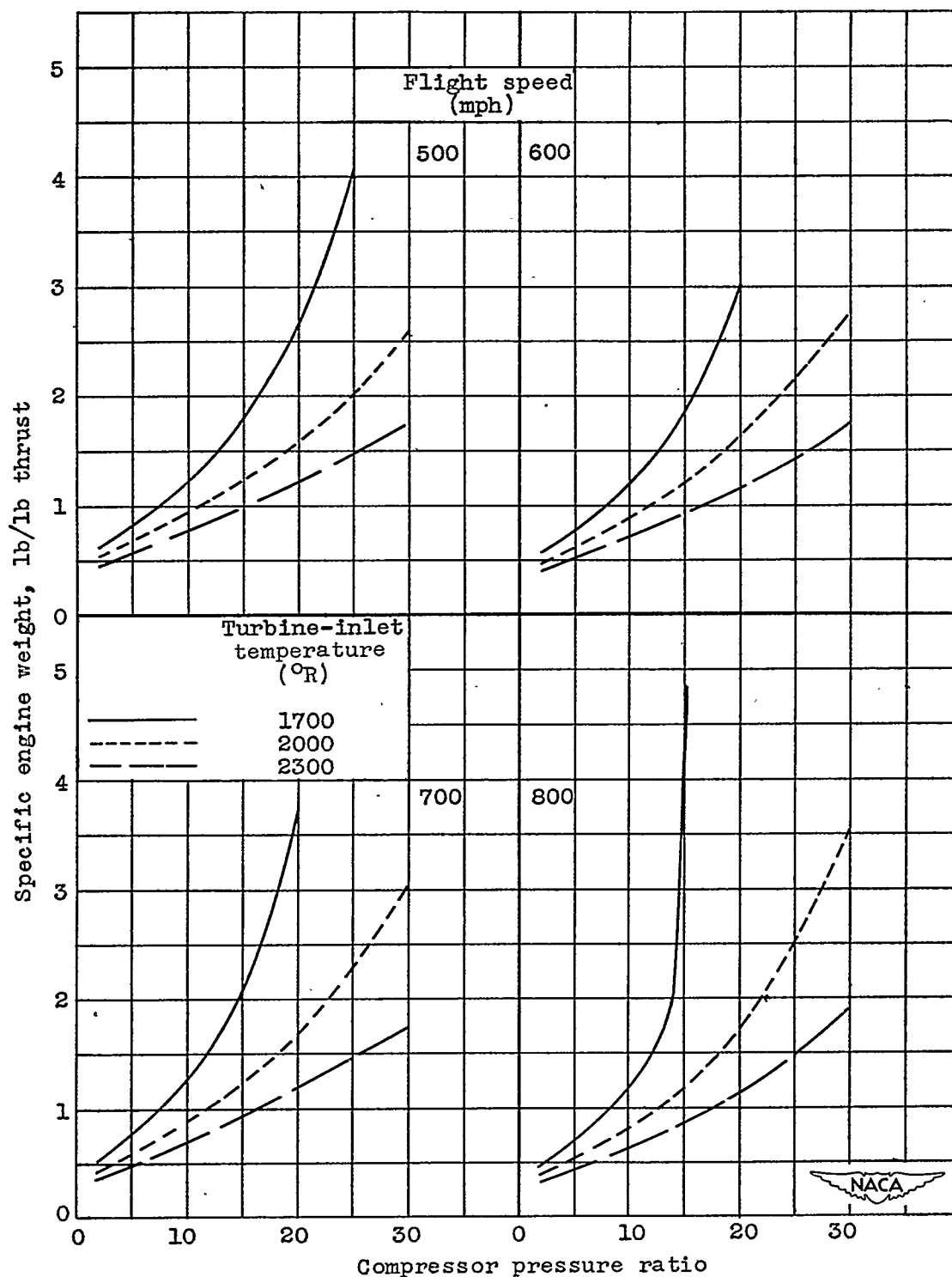
(d) Altitude, 70,000 feet.

Figure 6. - Concluded. Variation of net thrust per square foot engine frontal area with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



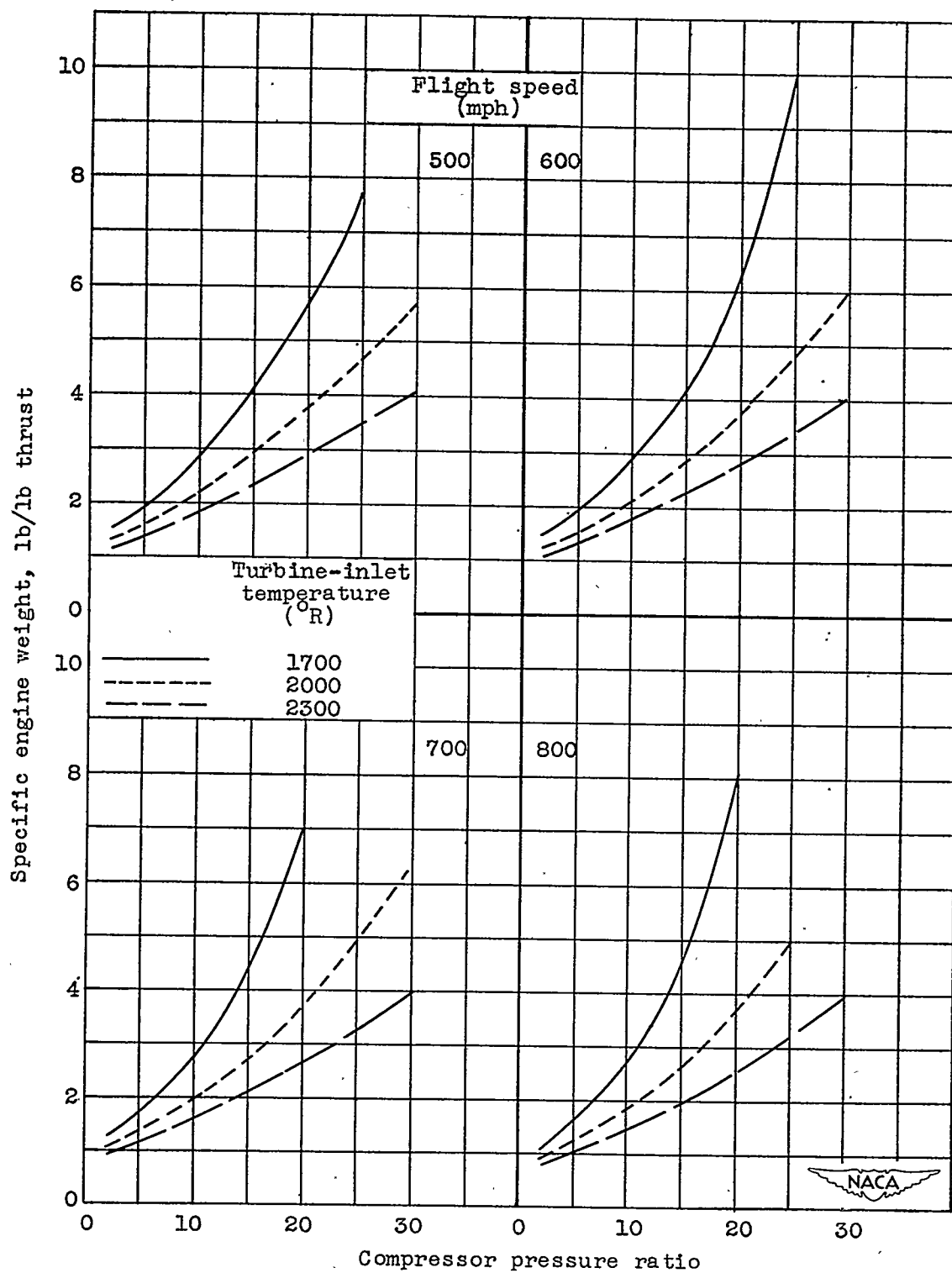
(a) Altitude, 10,000 feet.

Figure 7. - Variation of specific engine weight with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



(b) Altitude, 30,000 feet.

Figure 7. - Continued. Variation of specific engine weight with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



(c) Altitude, 50,000 feet.

Figure 7. - Continued. Variation of specific engine weight with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.

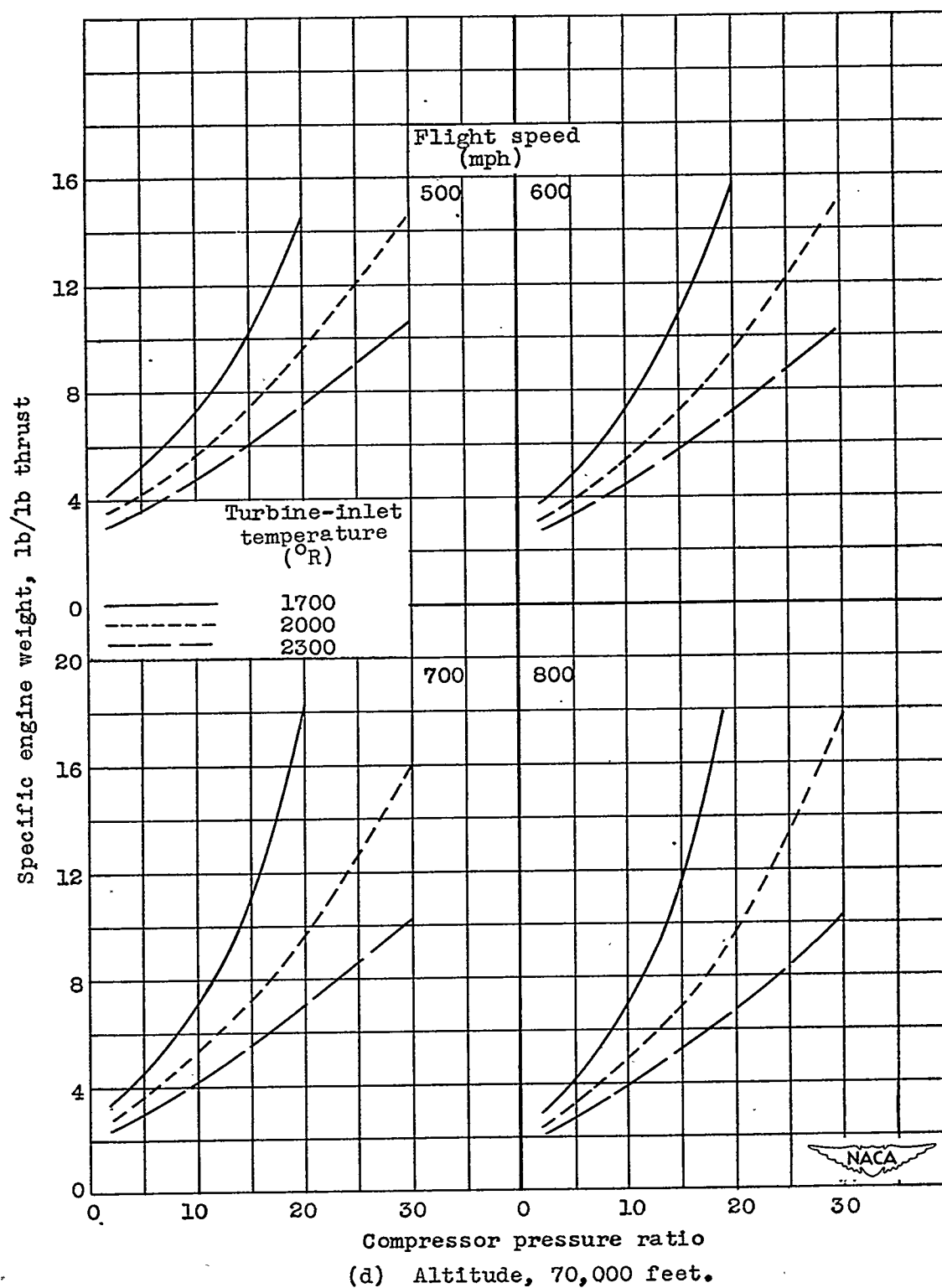


Figure 7. - Concluded. Variation of specific engine weight with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.

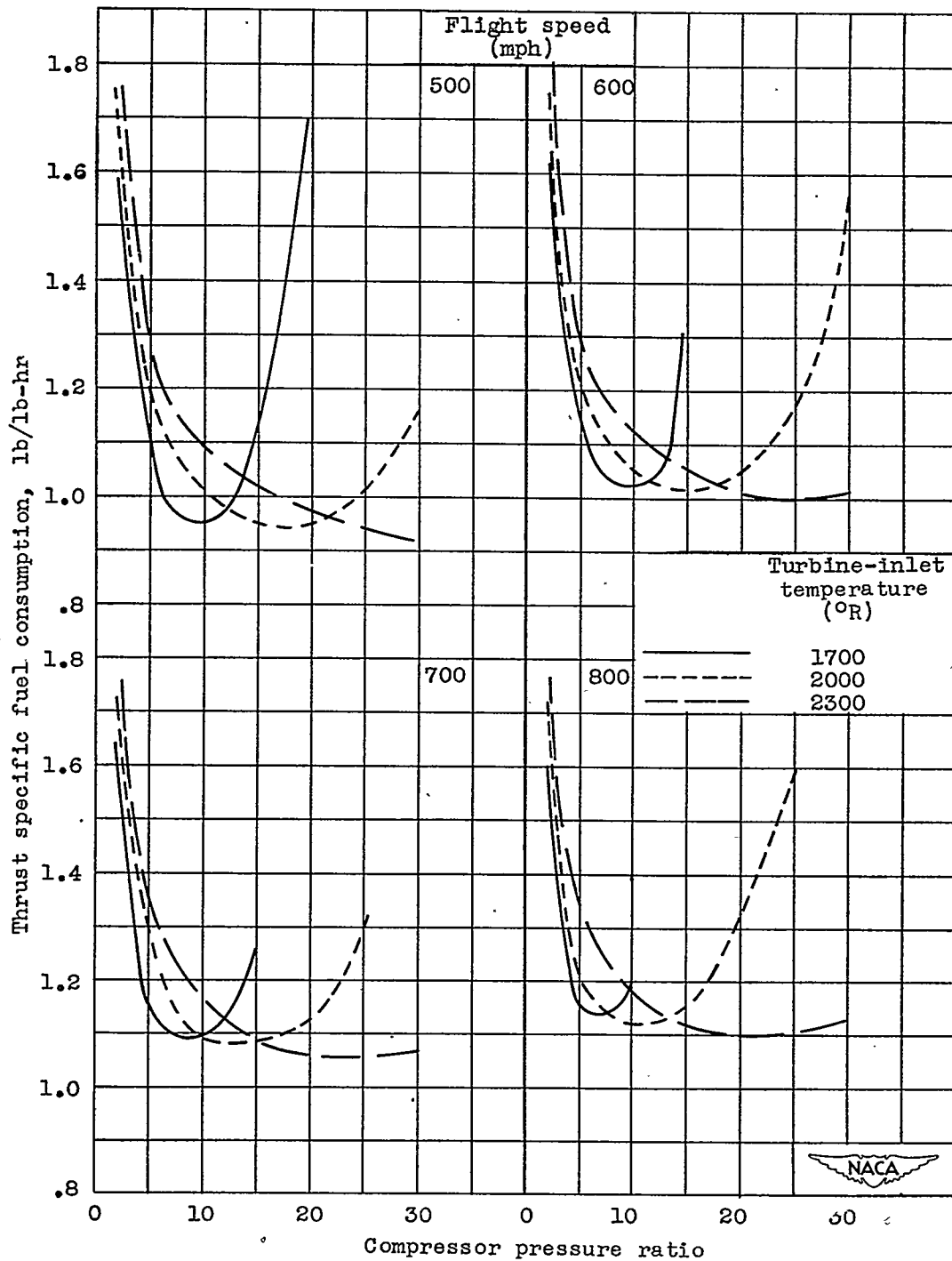
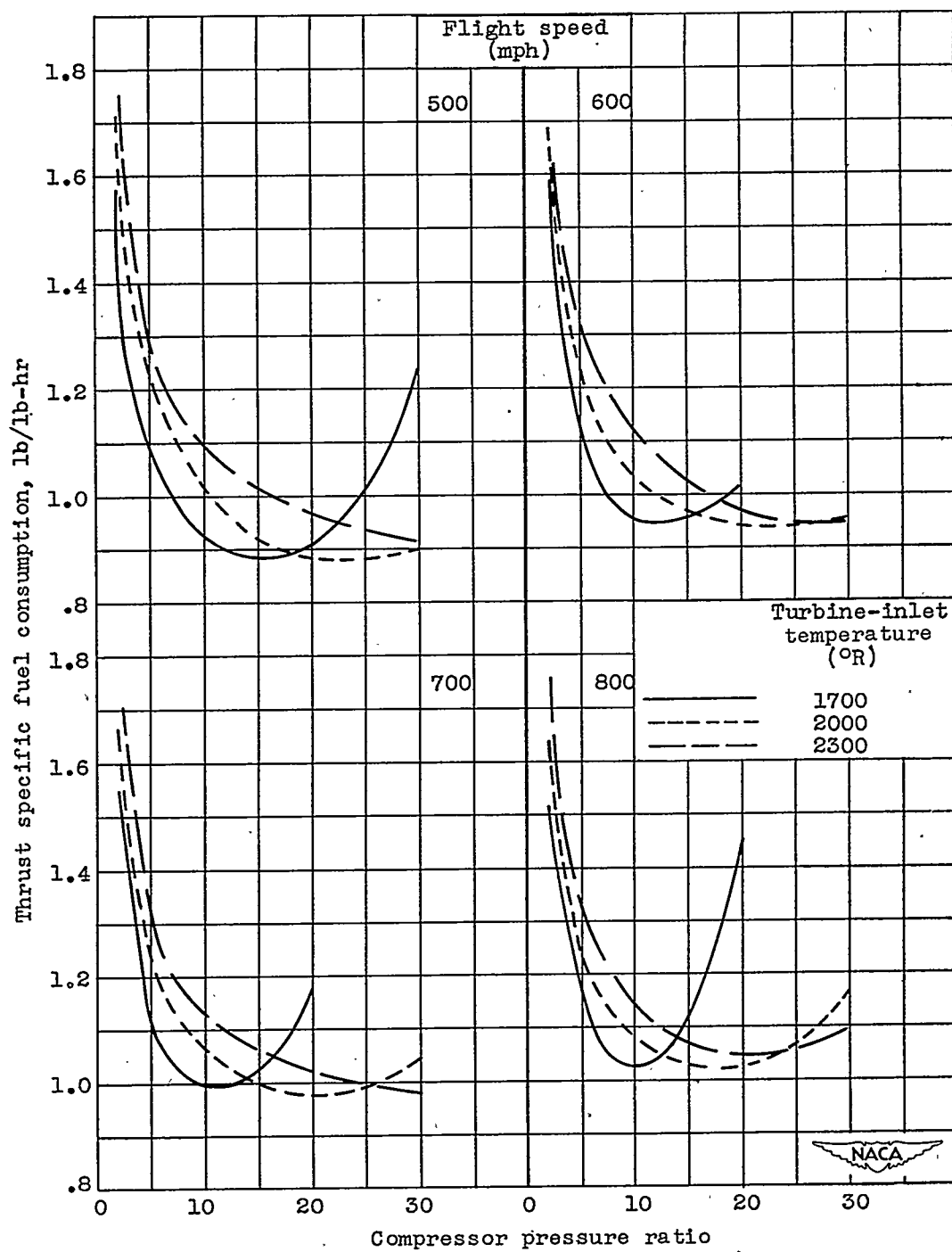


Figure 8. - Variation of thrust specific fuel consumption with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



(b) Altitude, 30,000 feet.

Figure 8. - Continued. Variation of thrust specific fuel consumption with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.

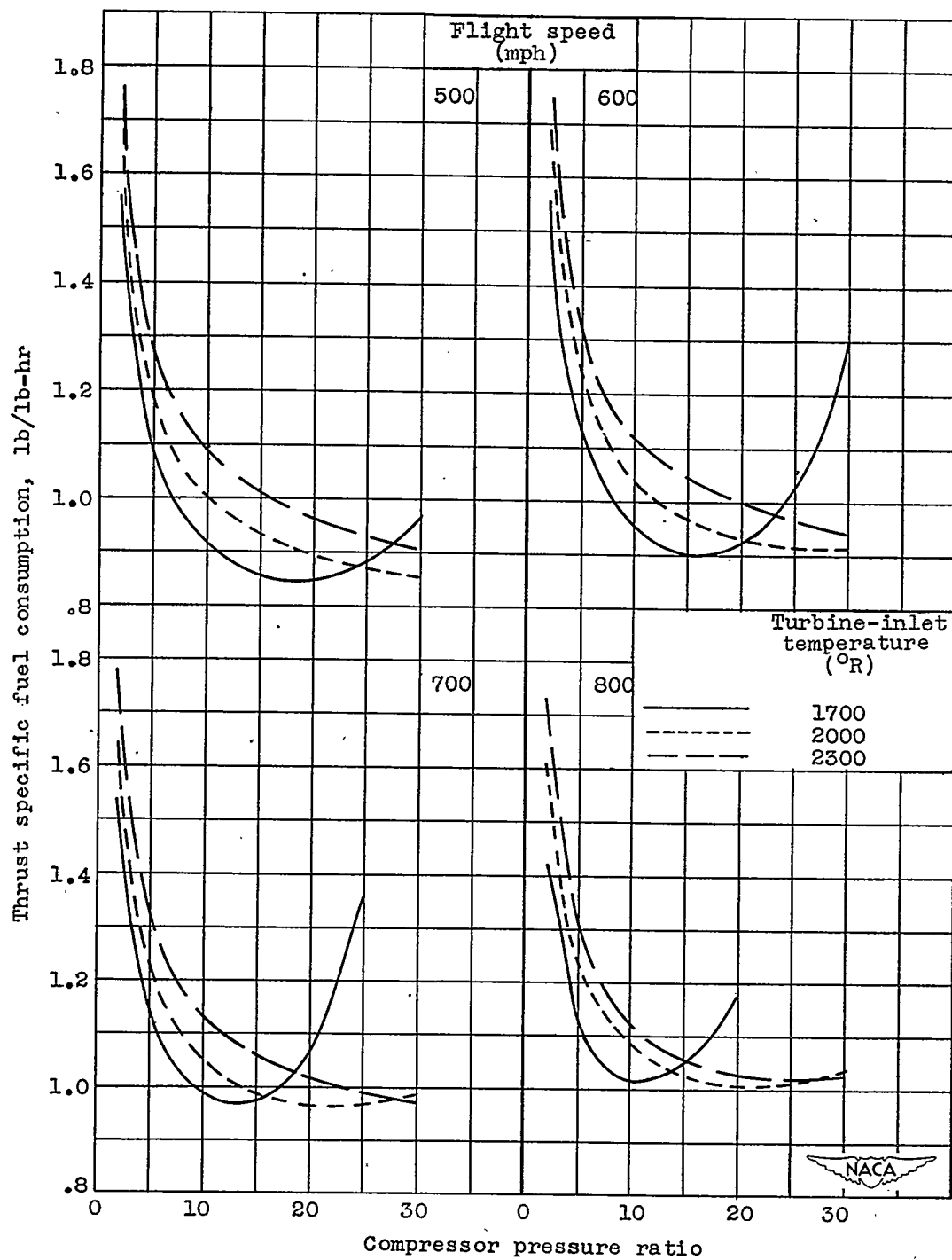
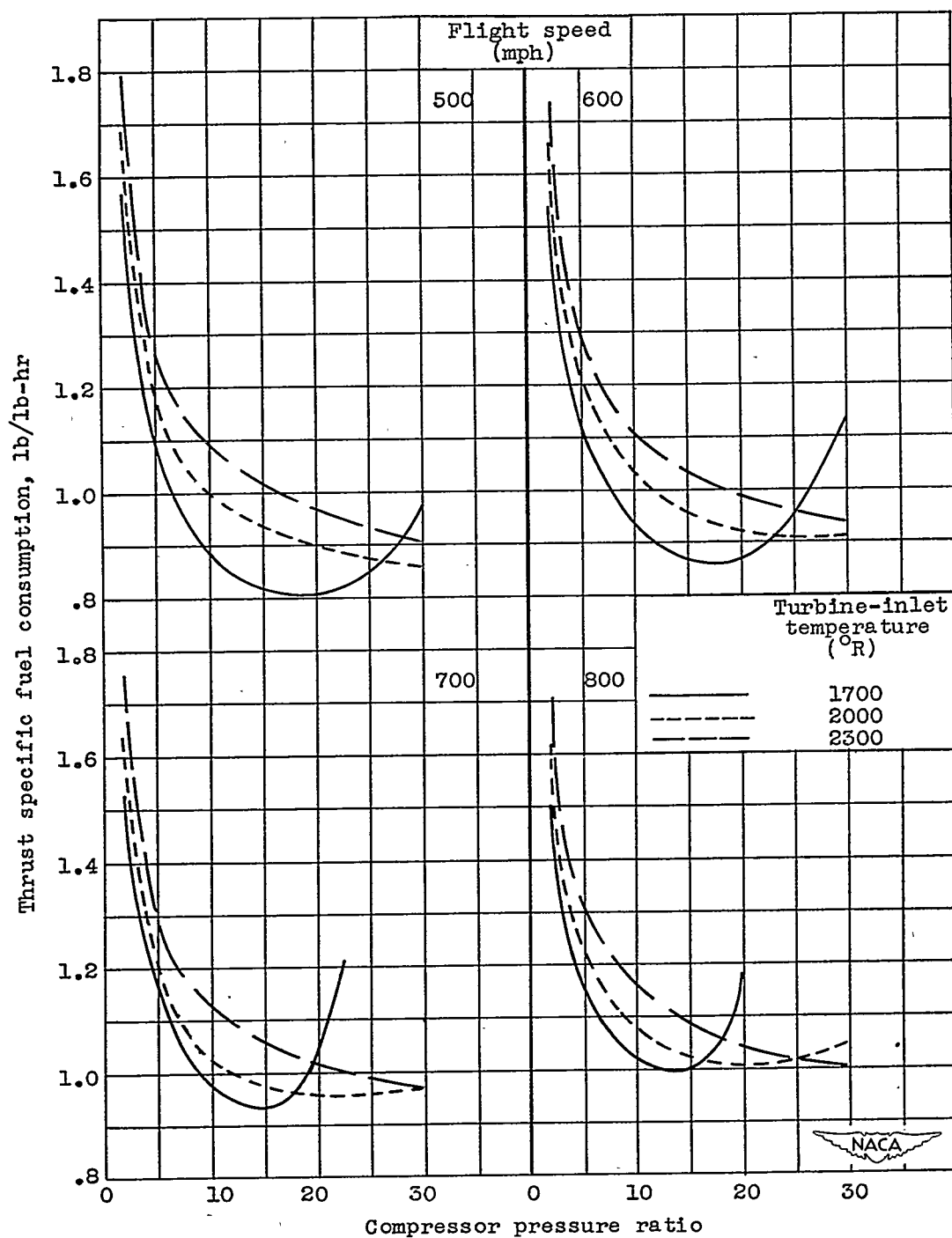


Figure 8. - Continued. Variation of thrust specific fuel consumption with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.



(d) Altitude, 70,000 feet.

Figure 8. - Concluded. Variation of thrust specific fuel consumption with compressor pressure ratio for various flight speeds and turbine-inlet temperatures.

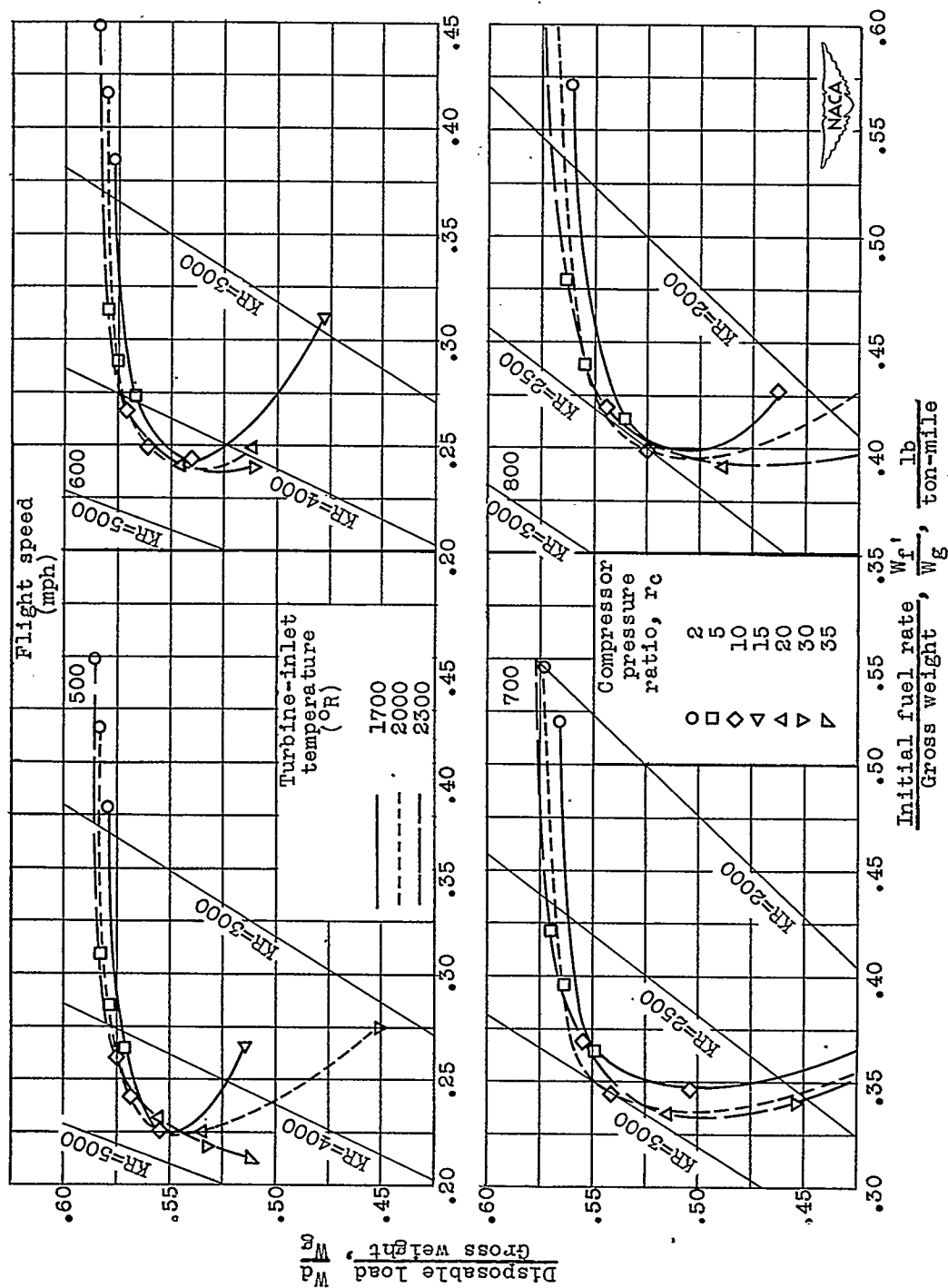


Figure 9. - Load-range characteristics of turbojet engine for various flight speeds, compressor pressure ratios, and turbine-inlet temperatures.

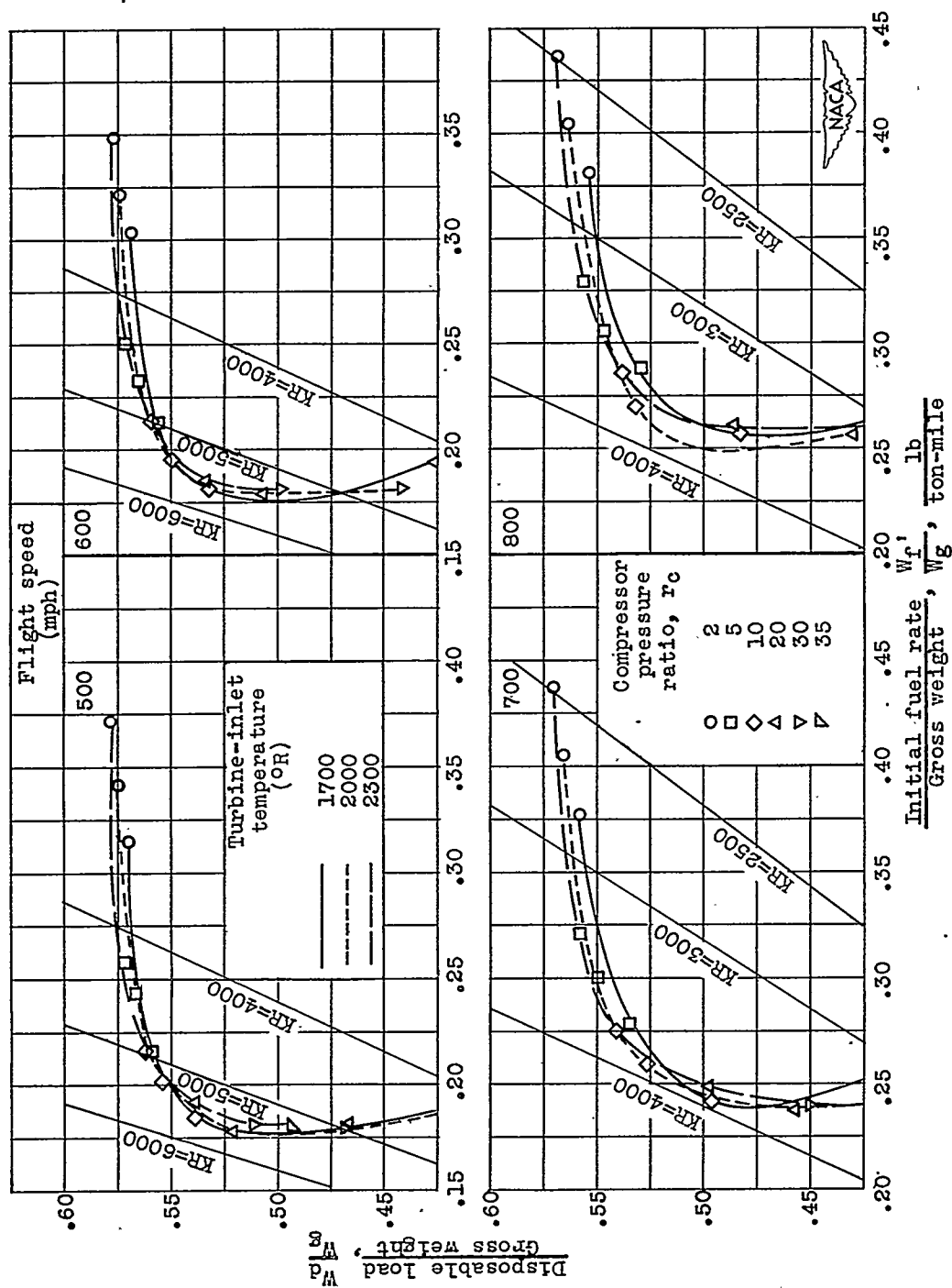


Figure 9. - Continued. Load-range characteristics of turbojet engine for various flight speeds, compressor pressure ratios, and turbine-inlet temperatures.

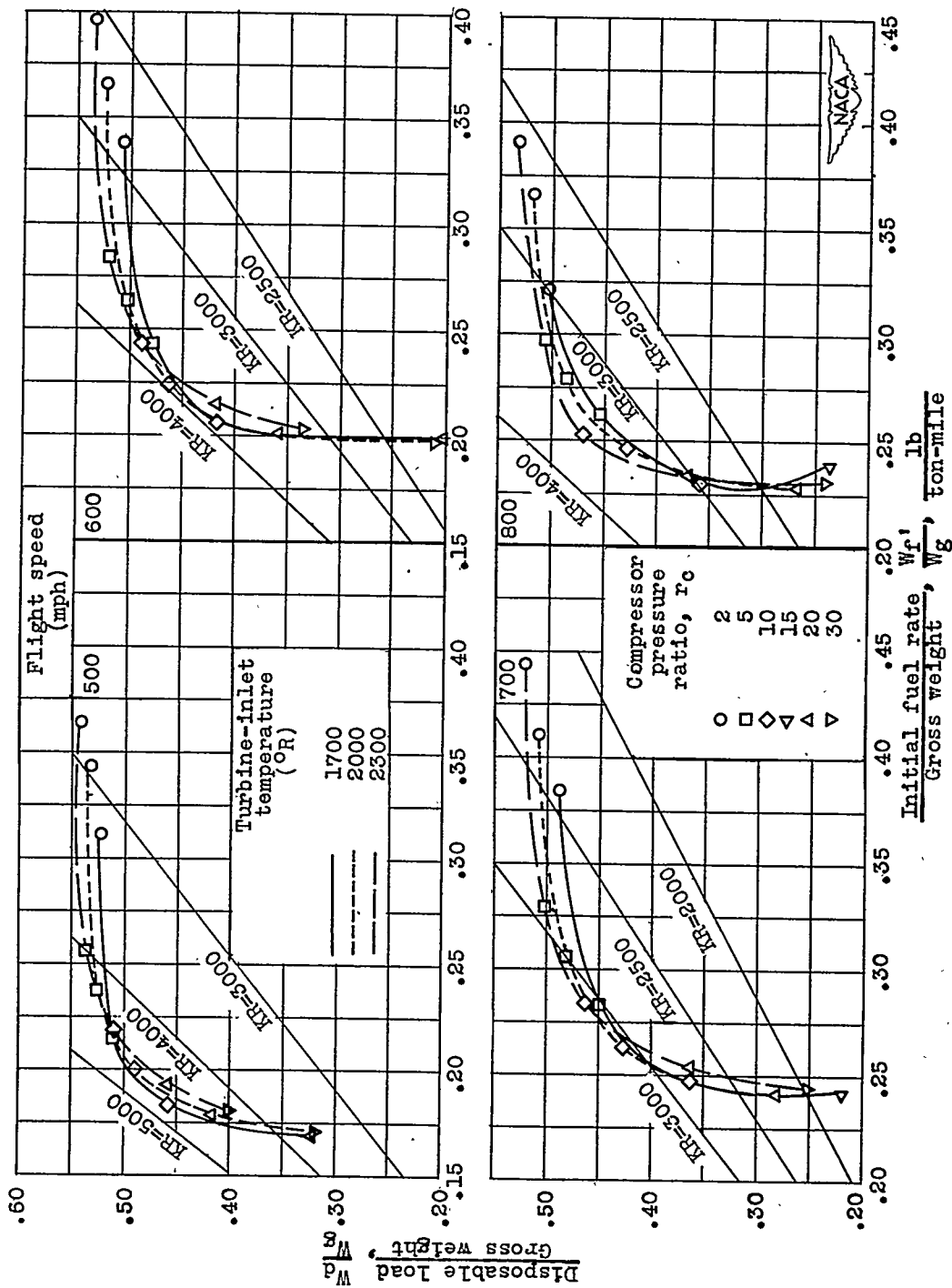
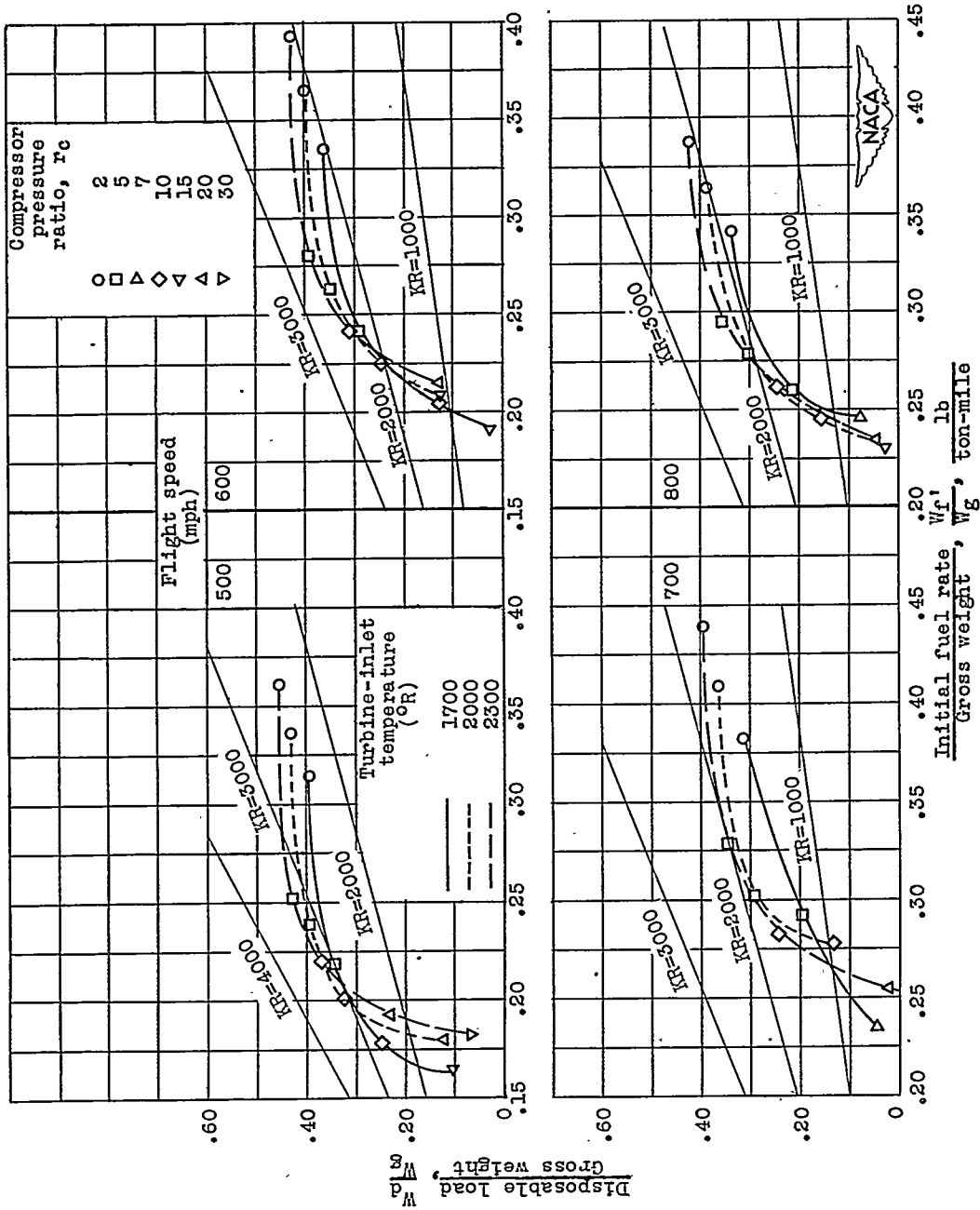


Figure 9. - Continued. Load-range characteristics of turbojet engine for various flight speeds, compressor pressure ratios, and turbine-inlet temperatures.



(d) Altitude, 70,000 feet.

Figure 9. - Concluded. Load-range characteristics of turbojet engine for various flight speeds, compressor pressure ratios, and turbine-inlet temperatures.

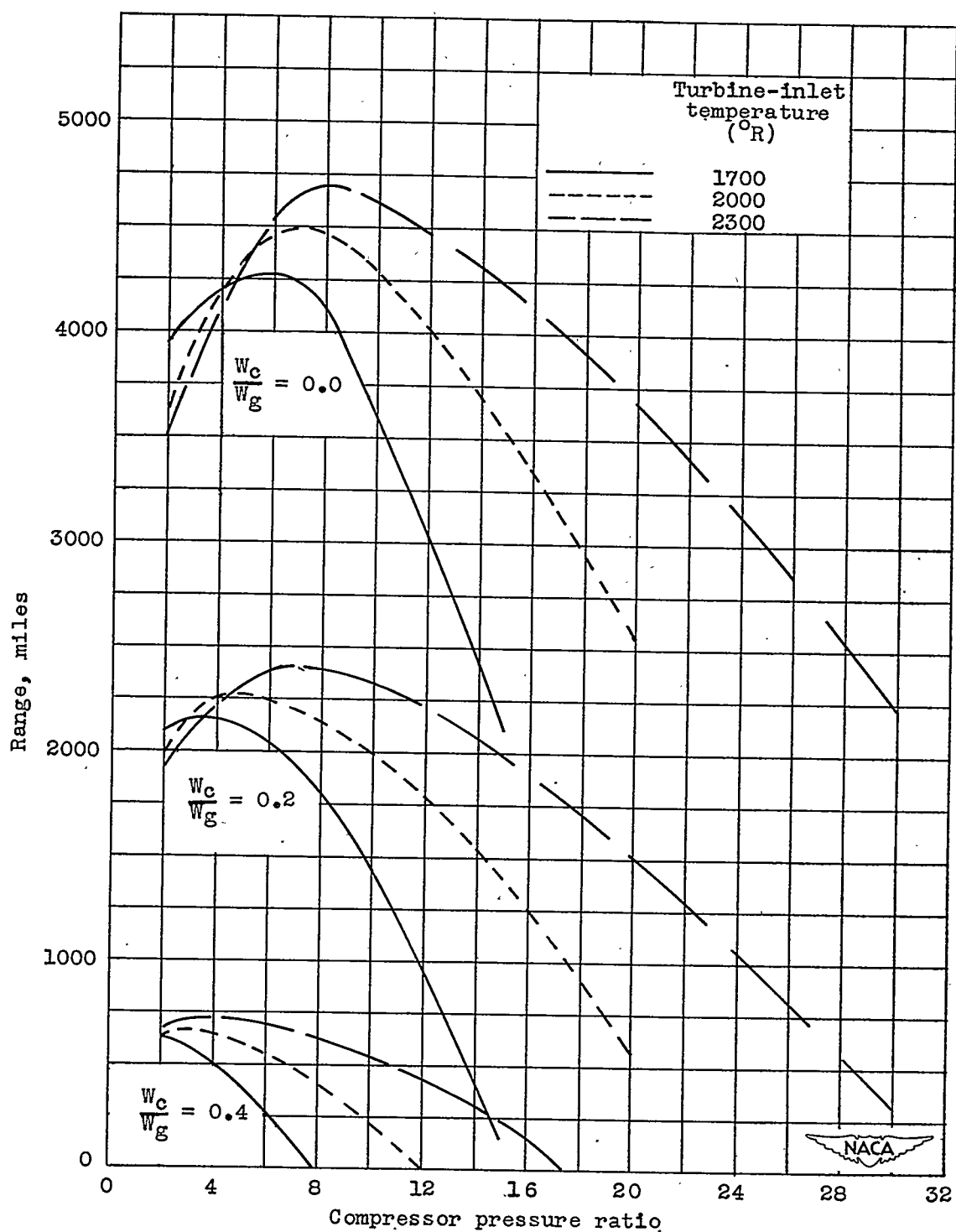


Figure 10. - Variation of range with compressor pressure ratio at 50,000 feet and 800 miles per hour for various pay loads and turbine-inlet temperatures.

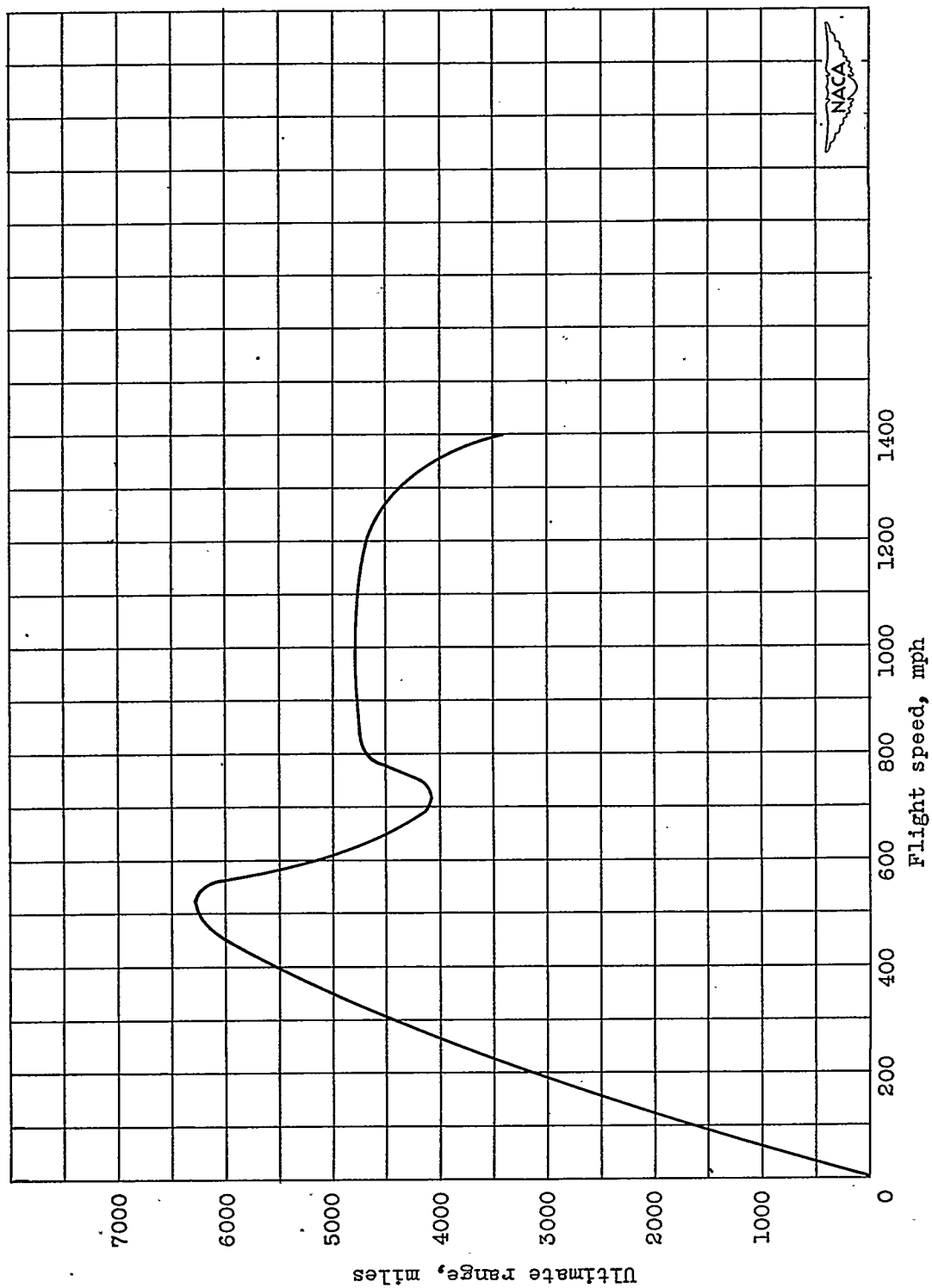


Figure 11. - Variation of maximum ultimate range of turbojet engine with flight speed. Altitude, 50,000 feet.